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ARTIFICIAL STABILITY INSTALLATION IN C-45 AIRPLANE

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CORNELL AERONAUTICAL LABORATORY, INC.

JULY 1953

WRIGHT AIR DEVELOPMENT CENTER

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NOTICES

WADC TECHNICAL REPORT 53-432

ARTIFICIAL STABILITY INSTALLATION IN C-45 AIRPLANE

Edwin A. Kidd
Arthur Gould

Cornell Aeronautical Laboratory, Inc.

July 1953

Directorate of Flight and All-Weather Testing
Contract No. AF-33(038)-21918
RDO No. 203-27

Wright Air Development Center
Air Research and Development Command
United States Air Force
Wright-Patterson Air Force Base, Ohio

FOREWORD

This report is a summary of work done by Cornell Aeronautical Laboratory, Inc., under Air Force Contract No. AF 33(038)-21918 in installing artificial stability equipment in a C-45 airplane. The report was released as Cornell Aeronautical Laboratory Report No. TB-754-F-2.

The contract was initiated under Research and Development Order No. 203-27, "Operational Evaluation of Improved Flight Path Stability and Control," and was administered under the direction of the Directorate of Flight and All-Weather Testing, Wright Air Development Center, with Mr. N. T. Wener acting as project engineer.

The subject C-45 airplane has been returned to the Directorate of Flight and All-Weather Testing. Presently it is employed as the prime device in a flight-test program which was undertaken to determine the effect of improved flight-path stability and control on the all-weather capability of an airplane.

ABSTRACT

A theoretical investigation is made into the effect of $C_{m_{DV}}$ on the airplane's longitudinal stability, and of C_{n_r}, $C_{n_{Dr}}$, $C_{n_{D\beta}}$, and C_{l_r} on the airplane's lateral stability. Finally, $C_{m_{Dv}}$, C_{l_r}, $C_{l_{Dp}}$, C_n, C_{n_r}, $C_{n_{D\beta}}$, and $C_{n_{Dr}}$ are provided artificially, or modified artificially, for the extensive variation of airplane handling characteristics.

Variable artificial force feel to controls is also provided.

The artificial stability system mechanical, electronic, and hydraulic components and installations are described.

Operation and performance of the artificial stability equipment is discussed, and a resume' of airplane handling characteristics with several combinations of artificial stability inputs is presented.

PUBLICATION REVIEW

This report has been reviewed and is approved for publication.

FOR THE COMMANDER:

Col USAF
for
H. B. MANSON, JR.
Colonel, USAF
Director of Flight and
All-Weather Testing

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS continued

LIST OF ILLUSTRATIONS continued

LIST OF TABLES

SYMBOLS AND CONVENTIONS

AXES

Stability axes are three dimensional right handed orthogonal axes fixed in the airplane at the beginning of a maneuver with their origin at the center of gravity of the airplane. The roll axis, X, and yaw axis, Z, lie in the plane of symmetry of the airplane with the X axis parallel to the free stream wind velocity. The pitch axis, Y, is perpendicular to the plane of symmetry.

X axis, positive forward

Y axis, positive to the right

Z axis, positive downwards

LINEAR DISPLACEMENTS

A linear displacement along a positive reference axis is considered positive.

ANGULAR DISPLACEMENT AND MOMENTS

An angular displacement or moment which is clockwise when viewed from the origin looking along a positive reference axis is positive.

VELOCITIES AND ACCELERATIONS

Velocities and accelerations, either linear or angular, are positive in the same sense as the corresponding displacement.

CONTROL SURFACE DEFLECTIONS

Positive elevator angle is associated with a downward movement of the elevator trailing edge. Positive aileron angle is associated with a downward movement of the right aileron trailing edge. Positive rudder angle is associated with a movement to the left of the rudder trailing edge.

SYMBOLS

b = wing span, ft.

C_L = $\dfrac{W}{Sq}$

C_l = airplane rolling moment coefficient

C_{l_p} = $\dfrac{\partial C_l}{\partial\left(\dfrac{pb}{2V_0}\right)}$

$C_{l_{Dp}}$ = $\dfrac{\partial C_l}{\partial D\left(\dfrac{pb}{2V_0}\right)}$

C_{l_r} = $\dfrac{\partial C_l}{\partial\left(\dfrac{rb}{2V_0}\right)}$

C_{l_β} = $\dfrac{\partial C_l}{\partial \beta}$

$C_{l_{\delta a}}$ = $\dfrac{\partial C_l}{\partial \delta_a}$

C_m = airplane pitching moment coefficient

C_{m_V} = $\dfrac{\partial C_m}{\partial\left(\dfrac{\Delta V}{V_0}\right)}$

$C_{m_{DV}}$ = $\dfrac{\partial C_m}{\partial D\left(\dfrac{\Delta V}{V_0}\right)}$

$C_{m_{\delta e}}$ = $\dfrac{\partial C_m}{\partial \delta_e}$

$C_{m_{D\theta}}$ = $\dfrac{\partial C_m}{\partial D\theta}$

C_n = airplane yawing moment coefficient

C_{n_p} = $\dfrac{\partial C_n}{\partial\left(\dfrac{pb}{2V_o}\right)}$

C_{n_r} = $\dfrac{\partial C_n}{\partial\left(\dfrac{rb}{2V_o}\right)}$

$C_{n_{Dr}}$ = $\dfrac{\partial C_n}{\partial D\left(\dfrac{rb}{2V_o}\right)}$

C_{n_β} = $\dfrac{\partial C_n}{\partial \beta}$

$C_{n_{D\beta}}$ = $\dfrac{\partial C_n}{\partial D\beta}$

$C_{n_{\delta a}}$ = $\dfrac{\partial C_n}{\partial \delta_a}$

D = $\tau \dfrac{d}{dt}$, 1/sec

g = Gravitational acceleration, ft./sec.^{2}

m = $\dfrac{W}{g}$

p = Airplane rolling angular velocity, rad/sec.

q = Dynamic pressure, psi.

r = Airplane yawing angular velocity, rad./sec.

S = Wing area, ft.^{2}

V = True airspeed, ft./sec.

W = Airplane gross weight, lbs.

β = Sideslip angle, degrees or radians

δ_a = Aileron deflection, degrees or radians

δ_e = Elevator deflection, degrees or radians

δ_r = Rudder deflection, degrees or radians

ρ = Mass density of air, slug/ft.^{3}

θ = Angle between X axis and horizontal, degrees or radians

τ = $\frac{m}{\rho S V_o}$, $1/sec$

superscript $\dot{x} = \frac{dx}{dt}$

subscript x_o = Initial condition at beginning of maneuver

SUMMARY

This report describes the function, operating procedure and maintenance of the mechanical, hydraulic and electronic installations in C-45F airplane, AF Serial No. 44-47546, which provide artificial stability to make all the natural modes of the airplane's motion essentially non-oscillatory and convergent. A brief summary of the theoretical analysis preceding the selection of the artificial stability inputs and a discussion of the stability provided as determined in flight tests is also given.

The artificial stability installation provides continuously variable artificial inputs proportional to yaw velocity, sideslip, rate of change of sideslip and yaw acceleration to the rudders; yaw velocity and roll acceleration to the ailerons; and rate of change of airspeed to the elevator. Artificial force feel on all three controls is provided with continuously variable force gradients.

This work has been accomplished under U.S. Air Force Contract AF 33(038) - 21918 for the All Weather Flying Section, Flight Test Division, Wright Air Development Center.

INTRODUCTION

The purpose of this contract was to incorporate such artificial stability into a C-45 airplane as to make all the natural modes of the airplane's motion non-oscillatory and convergent. Control forces were to be similar to those of the conventional airplane. Also, automatic turn coordination within a practical degree of accuracy was to be provided.

This contract has been accomplished by the Flight Research Department of Cornell Aeronautical Laboratory, Inc. at Buffalo, New York.

ARTIFICIAL STABILITY SYSTEM

Aerodynamic moments are applied via the three basic airplane control systems. The pilot is not mechanically connected to the surface controls and is therefore provided with artificial feel. Force feel and operation of the control surfaces is provided through electrohydraulic servomechanisms. The hydraulic servos operate at 1500 psi and are supplied by a hydraulic system installed for this purpose in the C-45 airplane. Electrical inputs to Moog servo transfer valves are supplied by a 400 cycle modulated carrier system of signal transfer with the power supply for all system components installed especially in the airplane.

Each pilot's control position is a function of pilot applied force/q and trim tab control position. Each control surface position is a function of the respective pilot's control position and particular airplane responses of position, rate and acceleration. Signals proceed from the various sensing devices to preamplifiers, gain controls and mixing amplifiers and thence to the servo transfer valves.

The servo actuators are mechanically connected to the airplane's control system. For operation through the normal controls, the pressure in the actuators is removed with suitable remotely operated valves. Operation of the airplane through the servos is possible only from the pilot's station with operation through the normal controls possible from the pilot's or co-pilot's station.

CONTRACT WORK ACCOMPLISHED

A theoretical study was made as reported in references (1) and (2) to determine the desired artificial inputs to provide the required damping of the modes of motion of the airplane. A brief flight test program of four flights was conducted to obtain the static stability characteristics of the C-45 airplane. With this test and theoretical information, the mechanical, hydraulic, and electronic installations were designed and fabricated. An effort was made to keep these installations as flexible as possible to allow for future modifications.

Control system modifications were made to mechanically separate the pilot from the normal airplane's control system. A complete 1500 psi hydraulic supply system for the electro-hydraulic servomechanisms was installed. All structural installations were stress analysed as reported in reference (3).

An extensive ground operational check-out was conducted and all artificial inputs and recorded items were calibrated. In nineteen flights of 41½ hours total flying time, the flight shakedown of the control system was accomplished, the amounts of the various artificial inputs to provide the desired damping were determined, and a brief pilot evaluation of the airplane's handling characteristics was obtained for various artificial stability configurations.

SCOPE OF REPORT

A description of the components of the artificial stability system as installed in the C-45 airplane is presented in this report. Operating instructions including maintenance and trouble-shooting information is provided. A list of all pertinent drawings and wiring diagrams is presented in Tables II and III.

A brief summary of the theoretical work preceding the installation is included as well as a summary of the handling characteristics of the airplane with artificial stability as determined in flight tests. Pertinent calibrations of the final installation are also included.

AIRPLANE DETAILS

An Air Force twin-engine Beechcraft C-45F, Serial No. 44-47546 was used for this project. A photograph of the test airplane is presented in Figure 1. Externally, the only modifications to the airplane are the sideslip vane boom on the right wing, the hydraulic oil cooler at the second cabin window on the right side of the fuselage, and installation doors in the horizontal stabilizer.

To provide space for components of the artificial stability control equipment, it was necessary to remove the nose fuel tank, the two left side cabin seats, and the forward right side cabin seat. Two seats remain on the right side of the cabin.

A three view dimensional drawing of the C-45 airplane is presented in Figure 2. The pertinent dimensions and details of the airplane are presented in Table I.

ANALYSIS AND DESIGN

THEORETICAL ANALYSIS OF LONGITUDINAL MOTION

The analysis of reference (1) for a C-45 airplane of gross weight = 8200 lbs., c.g. = 22.2% MAC, 5000 feet altitude, and true airspeed of 150 MPH with power off gives a longitudinal stability characteristic equation as follows:

$$\lambda^4 + 10.73\ \lambda^3 + 52.70\ \lambda^2 + 3.30\ \lambda + 4.33 = 0$$

Solution of the characteristic equation gives the following values for the roots:

$$\lambda_1, \lambda_2 = -.0232 \pm .287i$$

$$\lambda_3, \lambda_4 = -5.34 \pm 4.85i$$

These roots represent a phugoid oscillation with a period of 35.3 seconds and a time to damp to half altitude, $T_{1/2}$, of 48.1 seconds. The short period oscillation has a period of 2.09 seconds and $T_{1/2}$ of .210 seconds.

As the short period oscillation is quite highly damped no artificial stabilization of this mode was deemed necessary. The addition of artificial stability derivative $C_{m_{DV}}$ or pitching moment proportional to rate of change of airspeed was investigated to determine its effect on the phugoid motion. This effect on the roots of the characteristic equation is presented in Figure 3. For the analysed airplane configuration a value of $C_{m_{DV}}$ of approximately 1.3 is required to completely remove the phugoid oscillation. This correspods to an elevator motion in response to rate of change of airspeed, $\delta_e / \dot{V}$, of approximately -.82 deg./MPH/sec.

Figure 4 presents calculated time histories of velocity and normal acceleration responses to an elevator step deflection for the normal airplane or $C_{m_{DV}} = 0$, $C_{m_{DV}} = .5$, $C_{m_{DV}} = 1.3$. It was realized that the residual oscillation in airspeed with $C_{m_{DV}}$ added would be a function of the resolution obtainable with the $\dot{V}$- elevator servo system. This resolution was optimistically estimated to result in an uncontrolled oscillation of less than 1 MPH at 22.2% MAC. However, flight tests at a more aft c.g. indicated considerably greater oscillation in

airspeed. This problem is further discussed in the section, "Handling Characteristics with Artificial Stability".

To insure against excessive elevator response due to gust inputs of rate of change of airspeed, low pass filtering of the $C_{m_{\dot{V}}}$ input is required. To accomplish this the design of the airspeed differentiation circuit was such that it has a low natural frequency. Thus, its response to high frequency variations in differentiated airspeed is greatly attenuated.

THEORETICAL ANALYSIS OF LATERAL MOTION

The analysis of reference (1) for the same configuration indicated in the longitudinal analysis gives a lateral stability characteristic equation as follows:

$$\lambda^4 + 8.41 \lambda^3 + 25.04 \lambda^2 + 136.4 \lambda + 1.71 = 0$$

Solution of the characteristic equation gives the following values for the roots:

$$\lambda_1, \lambda_2 = -.45 \pm 4.24i$$

$$\lambda_3 = -7.495$$

$$\lambda_4 = -.015$$

These roots represent a Dutch roll oscillation of 2.39 seconds period and time to damp to half amplitude of 2.48 seconds; a rolling convergence of $T_{1/2} = .149$ seconds; and a spiral convergence of $T_{1/2} = 74.5$ seconds.

The effect on the roots of the characteristic equation of several artificial derivatives and artificially added increments to existing derivatives was investigated.

The effect of C_{n_r}, yawing moment due to yawing velocity, on the roots is presented in Figure 5. Artificially added C_{n_r} increases both the Dutch roll damping and spiral stability. The ΔC_{n_r} of approximately -1.0 required to damp

the Dutch roll oscillation completely is calculated to require approximately 1.4 degrees rudder per deg./sec. of yaw velocity. With this large amount of C_{n_r} however, control characteristics become considerably different from the normal airplane. The $\frac{pb}{2V_o}/\delta_a$ response to a step input as presented in Figure 10 indicates the resulting "automobile" type aileron control for $\Delta C_{n_r} = -1$. That is, ailerons must be held on in a steady turn.

The effect of $C_{n_{\dot{D}r}}$ or yawing moment proportional to yawing acceleration on the roots of the lateral characteristic equation is presented in Figure 6. The addition of this artificial derivative reduces the effective inertia in yaw and thus increases the damping in Dutch roll with no effect in the other modes of motion. Only about two-thirds of the amount of $C_{n_{Dr}}$ required for removal of the Dutch roll oscillation can be attained with the total rudder deflection available.

The effect of $C_{n_{D\beta}}$ or yawing moment proportional to rate of change of sideslip on the roots is presented in Figure 7. Removal of the Dutch roll oscillation can be attained through use of this artificial derivative without affecting the other modes of motion.

For spiral stability the effect of C_{ℓ_r} or rolling moment proportional to yaw velocity on the roots of the characteristic equation was investigated and is presented in Figure 8. The decrease in Dutch roll damping with increase in spiral stability was not considered serious, as the damping in Dutch roll could easily be replaced with one of the artificial derivatives discussed above. With the addition of relatively large amounts of C_{ℓ_r} the resulting initial roll out with rudder application was felt to be undesirable. This condition is demonstrated in Figure 10 in the $\frac{pb}{2V_o}/\delta_r$ response with ½ inertia removed and $C_{\ell_r} = -.40$. With the proper filter on this input, the roll response to yaw velocity input at the Dutch roll frequency can be attenuated and still provide the desired stability of the non-oscillatory spiral mode. The effect of a filter of time constant $T_\ell = .5$ is indicated in Figure 10.

It is necessary to attenuate all artificial derivative signals whose frequencies lie beyond the Dutch roll frequency to provide against excessive control motion resulting from gust inputs and/or noise inputs. Filter circuits described

in a later section of this report were used to accomplish this result.

ARTIFICIAL STABILITY TO BE PROVIDED

The original analysis of reference (1) resulted in a fairly well damped airplane with the following artificial configuration:

1. $C_{n_{Dr}}$ = .058 (50% inertia in yaw removed)

2. $C_{\ell_{Dp}}$ = .026 (50% inertia in roll removed)

3. C_{ℓ_r} = -.40, T_ℓ = .5

4. $C_{m_{DV}}$ = 1.3

The airplane responses to step inputs of aileron and rudder are presented in Figures 10 and 11 for this lateral configuration. The damping of the Dutch roll was better than that of the normal airplane, but still left much to be desired.

The additional analysis of reference (2) indicated the desirability of also providing $C_{n_{D\beta}}$ and C_{n_β} for Dutch roll damping and turn coordination respectively. C_{n_r} was provided, as considerable damping of the Dutch roll mode with this derivative could be realized with less C_{n_r} than that which results in changes in control characteristics.

With these four artificial derivatives and artificial modification to three of the airplane's inherent derivatives, a flexible installation was assured with means available to vary the airplane's handling characteristics rather extensively.

ARTIFICIAL FORCE FEEL

Irreversible hydraulic servomechanisms were employed to operate the control surfaces in response to the artificial stability inputs. Therefore, it was necessary to provide artificial force feel. A feel system was devised with hydraulic servos to position the pilot's cockpit controls in response to pilot-applied force.

This force feel could be supplied with the same servo that operates the control surfaces. However, the motion of the control surface in response to the artificial inputs would then be felt by the pilot. This configuration was not felt to be desirable; so it was decided to separate the pilot's controls from the control surfaces while operating with artificial stability and use two servos for each control system, one for force feel and one for control surface operation. The co-pilot's controls would remain connected to the control surfaces at all times.

This feel system was mocked up with a relatively small servo strut and tested. With strain gages on the servo strut, force gradients corresponding to airplane operation at low airspeed could not be attained without oscillatory instability. The phase lag introduced into the feedback loop by the inertia of the control column was sufficient to maintain an oscillation once it was excited by any rapid motion of the control column. With the use of a control wheel with strain gages on the wheel itself and correspondingly low outboard inertia sufficiently low force gradients were realized without oscillatory instability.

A discussion of the force feel characteristics of the present installation in the C-45 airplane is given in the section of this report, "Operation of the Artificial Stability System".

DYNAMIC PRESSURE DIVISION

To maintain the force feel of the normal airplane, division of all three control forces by dynamic pressure is necessary.

With the assumption of a relatively constant static margin, dC_m/dC_L over the speed range of the C-45 airplane a constant value of $C_{m_{DV}}$ is required to provide the same damping of the phugoid motion at all airspeeds.

$$C_{m_{DV}} = C_{m_\delta} \frac{d\delta_e}{dD\left(\frac{\Delta V}{V_0}\right)} = C_{m_\delta} \frac{d\delta_e}{d(\Delta^\circ V)} \frac{\rho S V^2}{m}$$

$$\frac{C_{m_{DV}}}{q} = C_{m_\delta} \frac{d\delta_e}{d(\Delta^\circ V)} \frac{2}{m}$$

Thus, the gain of the artificially added $\delta_e/(\Delta^\circ V)$ varies with airspeed squared or dynamic pressure to maintain a constant $C_{m_{DV}}$.

To maintain a constant value of the lateral derivatives $C_{\ell_{D_p}}$ and $C_{n_{D_r}}$ with airspeed, division by dynamic pressure is also necessary as:

$$C_{\ell_{D_p}} = C_{\ell_{\delta a}} \frac{d\delta_a}{d\dot{p}} \frac{2\rho S V^2}{mb}$$

$$\frac{C_{\ell_{D_p}}}{q} = C_{\ell_{\delta a}} \frac{d\delta_a}{d\dot{p}} \frac{4}{mb}$$

$$C_{n_{D_r}} = C_{n_{\delta r}} \frac{d\delta_r}{d\dot{r}} \frac{2\rho S V^2}{mb}$$

$$\frac{C_{n_{D_r}}}{q} = C_{n_{\delta r}} \frac{d\delta_r}{d\dot{r}} \frac{4}{mb}$$

MECHANICAL INSTALLATION

CONTROL SEPARATIONS

A mechanical control separation is provided on all three control systems to allow separation of the pilot's cockpit controls from the normal airplane's controls. Figures 12, 13 and 14 present schematic diagrams of the modified control systems. A list of pertinent detail and installation drawings is presented in Table II.

ELEVATOR CONTROL SEPARATION

Elevator control separation is accomplished by separating the torque tube connecting the pilot and co-pilot control columns under the cockpit floor. Figure 15 presents a photograph of this installation. A splined dog on one side of the torque tube separation mates with a female in a flange on the other side of the separation when in normal airplane control configuration. Operation of the control separation handle moves the splined dog away from the flange and separates the controls.

AILERON CONTROL SEPARATION

Aileron control separation and servo installations necessitated a bell crank installation in the left forward portion of the cabin. Figure 16 presents a photograph of the complete installation. A dog arrangement similar to that for the elevator separation is employed for separation of the controls. The bell crank to which the aileron surface cables are attached is fixed to the bell crank shaft; while the bell crank attached through cables to the pilot's control wheel is free to rotate on the shaft unless fixed with the splined dog.

RUDDER CONTROL SEPARATION

Rudder control separation is accomplished with bell cranks at the rudder control reduction pulleys under the floor of the center forward cabin. Figure 17 presents a photograph of this installation. The lower bell crank is fixed to the reduction pulleys and is attached to the co-pilot's rudder pedals. The upper

bell crank is attached only to the pilot's pedals. The lower bell crank is fixed to the bell crank shaft; while the upper bell crank is free to rotate on the shaft, unless fixed with the splined dog.

SERVO INSTALLATIONS

Each servo is installed as near to the item it controls as is practically possible. One difficulty experienced with all mechanical installations was the relatively light structure to which these installations had to be attached. It was necessary to stiffen the airplane structure in every case.

Each control surface servo is installed so that the servo strut reaches its internal stops just before the control surface reaches its own stops. This was done to prevent possible damage to the control system in the event of full servo strut travel when operating through the servos. Full travel of the control surfaces within the limits specified by the Erection and Maintenance Manual is still available with the servo struts attached. The force servos are installed so that the servo strut reaches its internal stops just before the respective surface servo.

The servo struts remain mechanically connected to the controls at all times. Removal of each strut is possible by removing the rod end bolt that attaches the strut to the bell crank or fitting. The rod end fittings on each servo strut are safetied with cotter keys to prevent rotation of the strut. As the feedback potentiometer wiper arm is attached to the servo strut with the case of the potentiometer attached to the strut housing, rotation of the strut could result in damage to the potentiometer.

ELEVATOR SERVOS

The elevator stick force servo is attached directly to the pilot's control column. Figure 18 presents a photograph of the installation on the left side of the cockpit.

The elevator surface servo is attached to the larger elevator cable, 3/16 inch diameter, through a bell crank. Figure 19 presents a photograph of this

installation just aft of the rear cabin bulkhead. The servo mount attachment to the floor channels has been reinforced since the photograph was made; details are presented in Drawing No. 359-004-10.

AILERON SERVOS

Figure 16 presents a photograph of the aileron wheel force and aileron surface servo installation in the left forward portion of the cabin. Existing pulleys in the aileron control system are maintained in this installation, although additional pulleys are installed.

The cables from the pilot's control wheel that were formerly attached to the aileron surface connecting cables are rerouted to the inboard bell crank. Cables from the outboard bell crank run to the aileron surface connecting cables at the attachment formerly used for the pilot's control wheel cables. A schematic of this control system modification is presented in Figure 13.

The outboard bell crank is made in two units. One unit with upper and lower arms has one cable and the surface servo strut attached. The offset lower arm has the other cable attached. As it was necessary to cross the cables to the outboard bell crank to maintain the proper direction of aileron surface deflection with control wheel deflection for operation through the normal controls, the offset bell crank was required to obtain the necessary clearances.

RUDDER SERVOS

The rudder pedal force servo is installed in the nose of the airplane, with the servo strut attached directly to the pilot's right rudder pedal. Figure 20 presents a photograph of this installation.

The rudder surface servo is installed in the right side of the horizontal stabilizer. It was necessary to cut three doors in the skin of the stabilizer for installation of the servo and bellcrank. The servo is attached through a bellcrank to the forward rudder bellcrank connecting cable. Figure 22 presents a photograph of the servo bell crank mount with the upper half of the bearing mount removed. A photograph of the servo installation is presented in Figure 21.

HYDRAULIC SYSTEM AND COMPONENTS

Hydraulic servo actuators with Moog transfer valves are used to provide force feel and control surface deflections. An engine driven hydraulic pump provides 1500 psi hydraulic supply to the transfer valves with system pressure regulated by relief valves and pressure fluctuations smoothed in a large system accumulator with small accumulators at the actuator installations where necessary. A system reservoir and a manually-operated pressure relief valve are provided. Solenoid-operated pressure valves and by-pass valves are installed at each servo actuator with pressure limiting check valves across the actuator pistons of the surface servo actuators.

A schematic of the entire hydraulic system is presented in Figure 23. Approximate locations of the various components are shown as well as the routing of the hydraulic lines.

SUPPLY SYSTEM COMPONENTS

A gear pump, Pesco Type 1P582CAC, is installed on the starboard engine. The vacuum pump on this engine was removed to allow installation of the hydraulic pump. With the mounting pad gear ratio of 1.5 to 1, the pump capacity is 3.6 gallons per minute at 1800 engine RPM.

System pressure is regulated with a Vickers Model AA11348A relief valve set to relieve at 1500 psi. An additional relief valve of the same type is set to relieve at 1650 psi in the event of a malfunction in the 1500 psi relief valve. The pressure cycle with an unloading valve instead of the relief valve proved unsatisfactory as the Moog servo transfer valve is somewhat sensitive to system pressure variations.

Use of the relief valve necessitated cooling of the hydraulic fluid. For in-flight cooling an eight inch Harrison Radiator oil cooler is provided in the return line just before the reservoir. The regulator on the cooler is set to by-pass all the hydraulic fluid through the cooler at a fluid temperature of approximately 130°F. Calculations indicate the ability of the cooler to maintain a fluid temperature of 130°F at 100 MPH with an outside temperature of 100°F. The location of this cooler may be seen in Figure 1. Ground operation of the hydraulic system with a hydraulic mule requires cooling if the system is operated longer

than approximately 15 minutes. Cooling may be accomplished with a simple heat exchanger using water as the coolant.

A reservoir of approximately 6 gallons operating capacity is installed in the cabin of the C-45 airplane. This reservoir is at atmospheric pressure and is vented overboard. Approximately 8 gallons of hydraulic fluid is required to fill the hydraulic system to the desired operating level.

A Purolater filter Type PR 306 with an AN 6235-3A-H filter element provides filtering of all impurities greater than 10 microns. In addition, filters are built into the Moog transfer valves for further protection. System filter elements should be inspected after every 10 hours of operation and replaced if dirty.

A 10 inch accumulator, AF type 4841-AA-1400 7B, is used to maintain constant pressure for the entire hydraulic system with additional 5 inch accumulators, AF type 4808-402961-0-1, at the elevator force, elevator surface, rudder force, and rudder surface servo installations. The aileron servos are only a short distance from the system accumulator, and an additional accumulator is not necessary. These accumulators are charged with approximately 600 psi pressure.

Hydraulic tubing for this installation was selected as per Table C, Section XIII of AN01-1A-8. The following sizes are used for the applications indicated:

APPLICATION	INSIDE DIAMETER	WALL THICKNESS	MATERIAL
Pressure line	3/8 in.	.049 in.	52SO
	*3/8 in.	.035 in.	61ST
Pressure line	1/2 in.	.065 in.	52SO
Drain line	1/2 in.	.035 in.	52SO

** 61ST tubing was used in some applications when 52SO was not available.*

MIL-H-5511 flex hose was used where flexible lines were required.

System pressure is indicated with a Type AN 5771-4, 0-2000 psi pressure gage. This is installed downstream from the relief valves.

A two-way selector valve, manually operated, provides for dumping of system pressure. This valve is connected directly from the main pressure line to the return line.

SERVO ACTUATORS

The hydraulic actuators are C.A.L. designed and fabricated. Total travel of the strut is ±3 inches with a Pacific Scientific Linear Motion Potentiometer attached to the strut to provide position feedback.

The piston "O" ring groove, made as per AN standards, is provided with a back up ring. The groove without a back up ring allowed movement of the "O" ring laterally in the groove. This movement resulted in a dead spot caused by effective backlash in the strut motion and was first noted in the force servos. The addition of a back up ring in the "O" ring groove corrected this difficulty.

The strut housing is of 24ST aluminum with ports drilled for direct mounting of the Moog transfer valve on the housing. Piston seals in either end of the housing with an "O" ring in the strut and one on the seal prevent leakage from the housing. The piston seals, held in with snap rings, also serve to limit the strut travel by acting as a stop for the piston face.

TRANSFER VALVE

The servo transfer valve for converting electrical input signals to hydraulic fluid flow is a Moog Model XIII-2 valve. A detailed description of this type of valve, its operation and performance is presented in reference (4). A schematic of the valve is presented in Figure 24.

For adjusting the null position of the valve, a screw adjustment is provided for balancing the cantilever spring load at one end of the spool against chamber pressure. This adjustment has been made on all servos in the C-45 installation and has not required additional adjustment over some three months of ground and flight operation. Balancing the spool in the null position is accomplished with pressure to the transfer valve and strut and the current to the transfer valve

disconnected. The spring load is adjusted until the servo strut remains in a fixed position or drifts very slowly to either end.

Current to the valve motor coil positions the flapper armature at the nozzle. The design of the hydraulic amplifier is such that the fluid developed force on the flapper is equal to the current force, and, as the nozzle area is a constant, the nozzle back pressure is then proportional to the current. This back pressure positions the valve spool which controls flow to the strut. With zero system pressure the valve spool is full over in one direction. The spool assumes a null position at pressures above approximately 1000 psi.

This type of transfer valve is designed to be relatively insensitive to variations in system pressure above 1000 psi. However, excursions in system pressure of ±100 psi which are typical of unloading valve regulation do result in small valve spool movements which are undesirable. As discussed earlier, this prohibited use of an unloading valve in the hydraulic supply system. To prevent excessive motion of the servo strut when pressure is applied or cut off to the servo transfer valve, it is necessary to have fast-acting selector valves on each servo. Later developed valves have two nozzles and are relatively insensitive to system pressure variations.

PRESSURE SUPPLY VALVES

To supply pressure to the servos three - port, two-position, solenoid operated selector valves are provided for each servo or set of servos. In the de-energized position, pressure is shut off to the transfer valve and the transfer valve is opened to drain. It is necessary to open the transfer valve to drain, so that pressure in the transfer valve is reduced rapidly to prevent a surge of flow into the servo strut as the spool of the transfer valve moves toward one end with the drop in pressure. Energized, the drain side is shut off, and pressure is supplied to the transfer valve. Thus, electrical failure cuts off the pressure supply.

Three four-way selector valves, modified to eliminate one port as indicated in Figure 25, are used for the aileron and rudder servos. Only one pressure supply valve is provided for the two aileron control servos; the aileron wheel

force and aileron surface servos are adjacent to each other, and use of the servos is always in pairs for a given control system. Two Saval selector valves, Part No. 23686, are used for the elevator force and surface servos. When more of these valves, or ones of similar type are available, they should be used to replace the modified valves in the aileron and rudder systems to improve reliability of operation.

BY-PASS VALVES

Three-port two-position, solenoid operated valves are provided to relieve pressure in the servo actuators in the event of pressure supply valve malfunction. These valves are C.A.L. designed and fabricated. Details of the valve are available in Drawing Nos. SH2045 through SH2047. De-energized, the by-pass valve opens each side of the actuator piston to drain. Energized, each side of the actuator piston is closed. With electrical system failure the valve fails safe, as pressure will be relieved from the actuator.

If a pressure supply valve malfunctions and does not close, pressure is relieved in the actuator by de-energizing the by-pass valve only if electrical current to the transfer valve is off. When an attempt is made to move the servo strut with feedback signal to the transfer valve, the flow through the transfer valve into the actuator is sufficiently large and the by-pass valve ports are small enough to result in a large pressure drop through the by-pass valve. This leaves a high working pressure in the actuator even with the by-pass valve open. With electrical current off to the transfer valve and consequently no feedback signal, the only flow through the transfer valve is leakage. This flow is small enough that the by-pass valve can handle it adequately. Hydraulic control system wiring has been made with this characteristic taken into account to insure the proper pressure relief.

To investigate any by-pass valve malfunction, disassembly and inspection of the valve is accomplished as follows:

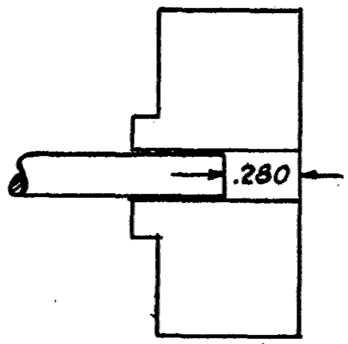
1. Remove the pre-loading spring from the end of the valve spool protruding from the valve body. This spring provides additional load to return the valve spool to the open position when de-

energized. Additional spring load is provided in the solenoid itself. Setting of the external spring load should be such that the solenoid will move the valve spool positively to the closed position and hold without oscillating. If the spring load is too great, the holding coil will not hold against it and the solenoid will oscillate with the starting coil cutting in and out.

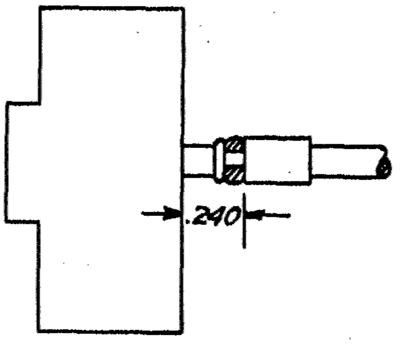
2. Remove the four (4) mounting screws that attach the solenoid to the valve body.

3. Pull the solenoid away from the valve thus removing the valve spool.

4. With proper adjustment the shoulder of the spool-solenoid coupling, Drawing SH-2045, will be no greater than .02 inch from the solenoid housing when the solenoid is energized. If it is apparent that the solenoid shaft has slipped in the coupling, unscrew the valve spool from the coupling. The solenoid shaft should be pressed and silver soldered into the coupling so the end of the shaft is .280 in. from the valve spool face of the coupling.



5. The valve spool should be screwed into the coupling and staked so that the dimension from the outer face of the "O" ring groove to the face of the coupling is .240 in.



6. If the "O" ring is worn, replace.

ACTUATOR PRESSURE LIMITING VALVES

As the hydraulic actuators were designed to provide the maximum required force for several applications, they have a maximum possible static force output of 1000 lbs. It is necessary to limit the force output to a value within the structural limits of the control system. Check valves with special springs are installed across the piston of the actuator to limit the maximum differential pressure in each direction. Pressure limiting is employed only on the control surface servo actuators. Pressure limiting is not required on the force servo actuators because the possible forces there will be limited by pilot effort, and the servo will resist any higher load internally at its stops. The force required for the pilot to overpower the force servos is excessive. With the check valves in the surface servo actuators the co-pilot can overpower the surface servos if necessary until pressure to the servos can be relieved with either of the emergency disengage switches.

The maximum force output of each surface servo actuator strut is limited to an equivalent force less than the limit, pilot-applied force used in design of the airplane's control system. These forces and equivalent differential pressures are as follows:

CONTROL SURFACE	APPROXIMATE LIMITING DIFFERENTIAL PRESSURE psi	EQUIVALENT ACTUATOR Force - lbs.	EQUIVALENT PILOT-APPLIED Force - lbs.
Elevator	395	263	93
Aileron	240	160	89
Rudder	215	143	147

Note: The maximum equivalent elevator stick force corresponds to 2.6 g with a center-of-gravity position of 25.9% MAC.

HYDRAULIC SYSTEM CONTROLS

Hydraulic system controls are provided so that either pilot or the test engineer may relieve pressure to the hydraulic actuators. A schematic of the control switch wiring is presented in Figure 26.

A panel of three switches is provided in the cockpit. Each switch controls current to the solenoids of the by-pass valves of a particular control system: elevator, aileron, and rudder. The test engineer's station in the cabin is provided with switches to control the pressure supply valves, by-pass valves and current to the servo amplifiers separately for each control system. Microswitches on the control disengage handles control current to the pressure supply valves and servo amplifiers. All switches to the same item are wired in series so any one will de-energize its particular solenoids or current relays.

Emergency disengage switches are provided on each control wheel in the cockpit. Operation of either of these switches opens the power feed circuits to all servo pressure and by-pass valves and amplifiers.

ELECTRONIC INSTALLATIONS

The electronic servo control in the C-45 airplane is designed to provide the required artificial stability inputs to the electrohydraulic servomechanisms as determined in the theoretical analysis. A high degree of flexibility has been incorporated into the control system to allow wide variations in the type and amount of artificial stability provided.

Pertinent airplane responses, control surface deflections and servo inputs are recorded on an oscillograph installed in the cabin of the airplane. Two instrumentation chassis located on the left hand side of the main cabin house the six servo control amplifier units, "q" divider, "disaster detector" and recording chassis. Figures 27 and 28 present photographs of this installation.

The control panel is located on the right hand side of the main cabin immediately aft of the hydraulic reservoir as indicated in Figure 29. A schematic identifying the various chassis and controls is presented in Figure 30.

In the forward right hand corner of the main cabin, the following sensing elements are mounted:

(a) Yaw Angular Accelerometer

(b) Roll Angular Accelerometer

(c) Yaw Velocity Gyro

(d) Roll Velocity Gyro

(e) Normal Accelerometer

Items a, b and c serve to provide artificial derivatives while at present the roll velocity gyro and vertical accelerometer output are recorded only.

Airspeed pressure pickups for recording dynamic pressure, for input to "q" divider and for input to airspeed differentiation are located in the nose section of the aircraft. These pickups are indicated in Figure 20.

The boom, located on the right hand wing, houses a sideslip vane to which are connected two synchros whose signal outputs are employed in providing artificial stability control signals proportional to sideslip and rate of change of

sideslip. A complete list of detailed wiring diagrams is presented in Table III.

POWER SUPPLY

A 400 cycle modulated carrier system of signal transfer is employed in the C-45 artificial stability system. Accordingly, the main source of operating power is a 28 volt D.C. input, 115 volt, 400 cycle, 750 volt ampere output inverter having incorporated integrally both A.C. voltage and frequency regulators. This inverter is located in the nose section of the airplane in space formerly occupied by the nose fuel tank (Figure 20). Rectified and filtered 400 cycle is used to supply amplifier tube plate potential. The heaters of all vacuum tubes are energized from the airplane's 28 volt D.C. source. Transformers are utilized to reduce the 115 volt supply to appropriate values for excitation of feedback potentiometers, force bridges and sensing pickups.

OPERATING POWER REQUIREMENTS

The power requirements of the servo control system including electrically operated hydraulic by-pass and pressure valves is approximately 1400 watts or 70 amperes at the nominal supply of 28 volts.

SERVO CIRCUITS - GENERAL

The six electronic servo control systems are in two general groups: surface servos and force servos. The servo control systems in each group are quite similar in construction and operation, differing primarily in number and type of inputs.

Each servo valve amplifier employs two stages, with a single triode voltage amplifier utilizing one half of a 12AT7 tube, transformer coupled to the push pull output stage which is a dual triode 12AU7. The input to this servo valve amplifier is a 6AU6 mixing or summation amplifier with a gain of approximately unity. The output stage functions as a differential cathode follower having the transfer valve coils connected in series with its cathodes, the two plates being tied together and supplied with 400 cycle A.C.

Servo position feedback is accomplished with a six inch linear motion potentiometer attached to, and moving simultaneously with, the servo actuating strut. This potentiometer is connected in a bridge circuit including another potentiometer which serves as the servo position control on the chassis panel (Figure 30). The moving arm of the linear motion potentiometer is connected to the input of the summation amplifier through an isolation cathode follower stage and gain control.

ELEVATOR SURFACE SERVO CONTROL

There are a total of three control signals fed to the elevator surface servo: elevator position feedback, position signal from the elevator force servo and one artificial stability input, differentiated airspeed. A schematic of the elevator surface servo control circuit is presented in Figure 31. Airspeed differentiation is accomplished by means of a velocity-servo system. A Sperry "Turn Control" unit is employed which consists of a two-phase induction motor coupled to a rate generator. The motor turns in response to a change in airspeed as measured by a syncrotel transmitter. The output of the motor driven rate generator is proportional to the rate of change of airspeed. A follow up synchro driven by the motor through a reduction gearing of 5700 to 1 nulls the signal from the syncrotel. This unit and its associated servo amplifier are located on the elevator surface servo control chassis.

The airspeed sensing unit is a Kollsman synchrotel or pressure actuated synchro having a range of 40-300 miles per hour with a rotation of one degree per mile per hour change of airspeed. This unit is one of the three airspeed pickups located in the nose section of the airplane.

The rate generator output or differentiated airspeed signal is connected through a pre set and locked sensitivity control potentiometer on the panel of the elevator surface control servo. Output of the sensitivity control is amplified by a Kay Lab 106A amplifier and a divider driving amplifier. The signal after division by q is fed to the gain control and then to the elevator surface servo control mixer.

To limit the frequency response of the differentiated airspeed input and

consequently the response of the airplane to gusts, damping is provided in the differentiated airspeed signal channel. This is accomplished by connecting a potentiometer across the rate generator output and feeding back a small amount of rate signal to the mixer input of the differentiator servo amplifier. The potentiometer is designated "damping" and is located on the elevator surface servo control panel.

AILERON SURFACE SERVO CONTROL

There are a total of four signals fed to the aileron surface servo. In addition to the surface position feedback and the force servo position signals, there are two artificial control signals: yaw velocity and rolling angular acceleration. A schematic of this control circuit is presented in Figure 32.

The yaw velocity gyro is a Doelcam model K, having a full scale rate of 60 deg/sec., damping ratio of .7 critical and an undamped natural frequency of 28 cycles/second. Excitation of both the gyro motor and microsyn pickoff is provided by a 115 to 26 volt step down transformer located in the power and rudder servo control chassis. The thermostatically controlled gyro heater is supplied with 28 volts D.C. from the ship's supply. The signal from the rate gyro microsyn pickoff is amplified by a driver amplifier having an approximate gain of four and is then filtered.

To provide the desired phase lag in the yaw velocity control signal fed to the aileron for spiral stability, a filter of rather large time constant was required. The filter comprises a single order resistance capacitance network with four values of resistance available for adjustment of the time constant. The time constant selector switch is located on the q divider chassis panel. A schematic of this filter is presented in Figure 32, and a plot of amplitude ratio and phase angle versus frequency is found in Figure 36.

The roll angular accelerometer is a Statham strain gage bridge circuit type AA5 having a range of ±3 radians/second squared, damping ratio of .7 critical, and an undamped natural frequency of 7 cyc./sec. Bridge balance potentiometers for both resistance and phase are located on the aileron surface servo control panel. Amplification of the roll angular acceleration signal is accomplished

with a Kay Lab 106A pre-amplifier and a driver amplifier after which the signal is divided by q. Filtering is accomplished with a filter having a nominal corner frequency of one cycle/second. This filter is described below under "Control Signal Filters". After being divided by q and filtered, both yaw velocity and roll acceleration signals go through their respective gain controls and to the aileron surface servo mixer amplifier.

RUDDER SURFACE SERVO CONTROL

A total of six signals are fed to the rudder surface servo. In addition to the surface position feedback and position signal from the rudder force servo, there are four artificial control signals: sideslip, differentiated sideslip, yaw angular acceleration and yaw velocity. A schematic of this control circuit is presented in Figure 33.

The rudder surface servo valve amplifier has in its first stage a 12AU7 tube instead of a 12AT7, as in the other servo valve amplifiers. This low gain tube was installed during the ground operation check-out phase in an effort to eliminate an oscillatory buzz in the rudder surface servo. The resulting reduced frequency response of this servo as discussed in a later section, "Component Performance", has not presented any difficulties. Later investigation indicated this buzz was probably attributable to the check valves across the actuator piston; as the valves were adjusted and the buzz eliminated, replacement of the low gain tube with a 12AT7 should not reintroduce this difficulty.

The sideslip and differentiated sideslip sensing synchros are located in the boom on the right hand wing and are excited from a 26 Volt 400 cycle transformer located in the power and rudder control chassis. Nulling synchros for both circuits are provided on the panel of this chassis. The output of the sideslip nulling synchro is filtered using the filter of Figure 37, after which the signal is connected to the sideslip-yaw velocity selector switch located on the panel of the power and rudder chassis. The operation of this switch may best be seen by reference to Figure 33. It will be noted that a single gain control serves to control either the sideslip or yaw velocity to rudder signal.

The output of the sideslip nulling synchro which provides input to the differentiator is amplified by means of a V.T.C. type A21 step up transformer and then the signal is demodulated using a 400 cycle chopper. Filtering of the signal is provided by an 8 microfarad shunt capacitor immediately prior to differentiating by the RC differentiating network, which comprises a 1.5 microfarad series capacitor and a 5000 ohm shunt resistor. After differentiation, the carrier is reinjected using a second 400 cycle chopper. Amplification of the signal by means of a Kay Lab 106A pre-amplifier and driver amplifier is provided prior to division of the signal by q. After division, the signal is filtered using the one cycle corner frequency filter (Fig. 37) before being fed to the gain control and rudder surface servo mixer input.

Inasmuch as yaw velocity is used as an artificial control input to both the aileron and rudder surfaces, the signal used for rudder control is picked up at the point where it enters the divider chassis. Additional amplification is provided with a step up transformer before filtering with the one cycle corner frequency filter (Figure 37). The filtered signal then is connected to the previously described sideslip-yaw velocity selector switch.

FORCE SERVO CONTROLS

Each force servo has as inputs to its mixer amplifier: position feedback, force signal from the strain gage bridge, and trim control signal. The elevator and aileron force servo amplifiers and associated circuits are located on one chassis. Figure 34 presents a schematic of these control circuits. The rudder force servo amplifiers and associated circuits are located in the rudder force and surface servo control chassis. A schematic of this control is presented in Figure 35.

A special control force wheel as furnished by the Air Force replaces the normal pilot's control wheel. Strain gages on the forward and aft sides of the control wheel arms are connected as a four arm bridge whose output is proportional to applied elevator force. For aileron force, strain gages are installed on the upper and lower sides of the control wheel arms. Phase and resistance balance potentiometers for the aileron and elevator force bridges are located on the chassis.

Rudder pedal force attachments as furnished by the Air Force are installed on the pilot's normal rudder pedals. The four arm strain gage bridges of each pedal are connected in parallel and have excitation balancing potentiometers in a junction box under the cockpit floor. These were balanced on installation and do not need adjusting. Bridge phase and resistance balance potentiometers are located in the amplifier chassis.

Each force servo has a positioning potentiometer on its respective chassis. Force bridge outputs are amplified by a Kay Lab 104A amplifier and driver amplifiers and then divided by dynamic pressure. Force gain controls connected between the divider output and the force servo mixer amplifier are located on the test engineer's control panel as are feedback gain controls.

CONTROL SIGNAL FILTERS

The airplane motions to be controlled with the artificial stability inputs are at frequencies no greater than approximately .35 cyc./sec. To reduce the effect of gust inputs all artificial stability items are filtered at least to the extent of the filter indicated in Figure 37. These filters have not prevented attainment of the desired amounts of damping of the airplane motions.

Resistance capacitive network filters which limit the amplitude response to approximately 0.1 at 10 cycles/second are used in the differentiated sideslip, sideslip, rolling acceleration, yawing acceleration and yaw velocity to the rudder control signal channels. Schematics of this filter are presented in each control circuit schematic and a plot of amplitude and phase response versus frequency is shown in Figure 37. The above filters are inserted in the circuit between the q divider and the servo amplifier input and are located in the divider chassis. In each of the control signal filters the carrier is demodulated using a 400 cycle chopper, filtered, and the carrier reinjected using a second chopper. The transformers at the output of the filters serve a dual purpose: first they provide a gain of approximately 1.5/1 through the filter and secondly, they are broadly tuned to approximately 400 cycles which provides a measure of wave shaping to the injected carrier.

In addition, a filter with a variable time constant is located in the divider chassis. This is used to filter the yawing velocity signal fed to the aileron surface servo. A description of this filter will be found in the section entitled, "Aileron Surface Control Servo".

RECORDING FILTERS

Inductive-capacitive filters are provided on all recorded channels. Circuits of these filters are shown on the servo control circuit schematics and frequency responses are presented in Figure 38.

DYNAMIC PRESSURE DIVIDER

Division by dynamic pressure is accomplished with two banks of potentiometers, a total of nine, which are driven through 1:1 gearing by the divider servo motor whose position is proportional to q. One of the driven potentiometers in conjunction with the airspeed pressure pickup forms a bridge, the output of which excites the divider servo amplifier. The driven potentiometer therefore acts to null the bridge error or excitation signal caused by excursions of airspeed.

The ganged potentiometers connected in the force and artificial derivative circuits act to vary the signals in conformance to changes of dynamic pressure. Figure 39 presents a schematic of the dynamic pressure divider. Note that the potentiometers are shunted by a resistance load which provides the 1/q function required.

A divider sensitivity control is located on the panel of the divider chassis unit. Adjustment of this control may be conveniently performed on the ground. If an airspeed change of sufficient magnitude be made to effect divider operation damping may be determined by visually observing the potentiometer drive gearing. The proper setting of the sensitivity control is the highest that will prevent hunting of the servo motor. This control has been set to the proper value and normally will not require readjustment.

TRIM CONTROL

Electrical trim control is provided for each servo system and is operated with the normal airplane trim control for the particular control surface. A ten turn potentiometer, chain or wire belt driven from the airplane trim control wheel, is arranged in a bridge circuit with two 5000 ohm resistors. The bridge output is fed to the force servo mixer amplifier. Schematics of this servo input are presented in Figures 34 and 35 for the respective force servos. The excitation transformer for the trim circuits is located in a junction box on the aft left side of bulkhead number five. The gain of these trim control inputs may be varied by adjustment of the potentiometers on the trim junction box. These gains have been set to values that provide trim characteristics that are approximately the same as those of the conventional C-45.

DISASTER DETECTOR

To prevent hard over signals in the event of electrical malfunction, a "disaster detector" is incorporated in the elevator and rudder servo control systems.

A 12AT7 tube is utilized as the detector with its grids connected across the servo transfer valve (cathodes of the servo valve amplifier output tube). A sensitive relay having a normally closed set of contacts is inserted in each of the plate circuits of the detector tube. The two normally closed sets of contacts are connected in series and control operation of a relay in the power circuit of the particular servo system. Inasmuch as the detector tube grids are at the same potential as that across the transfer valve coils, it is evident that the control tube bias will vary in accordance with valve current or change of potential across the transfer valve coils. Sensitivity of the detector is adjusted by means of potentiometers connected in the detector tube cathode circuits. Detector tube bias is normally adjusted so that excursions of voltage across the transfer valve coils sufficient to permit nearly full surface travel from mid-strut position will not open the servo circuit. However, any rise over and above this value will cause the disaster detector to operate to close the pressure supply valve, open the by-pass valve, and cut off the current to the particular servo amplifier.

Hydraulic malfunction resulting in a hard over actuator strut that might be experienced with transfer valve failure is also controlled by the disaster detector. This type of malfunction will provide a large feedback signal that can not be nulled out and this signal will operate the disaster detector.

RECORDING CIRCUITS

Items recorded in the C-45 include the following:

1. Aileron surface position
2. Elevator surface position
3. Rudder surface position
4. Sideslip
5. Indicated airspeed (dynamic pressure)
6. Normal acceleration
7. Rolling angular acceleration
8. Yawing velocity
9. Rolling velocity
10. Yawing angular acceleration
11. Differentiated sideslip
12. Differentiated airspeed

The recording medium is a 24 channel Heiland oscillograph.

Recording circuits for items 1-6 are located on the recording control chassis with nulling synchros and step attenuators for the three surface positions and sideslip channels on the chassis panel. Bridge balance potentiometers and step attenuators for the indicated airspeed and vertical acceleration channels are also on this chassis panel.

Position measuring synchros are located as follows:

Aileron - at surface push rod in right hand wing

Elevator - in tail cone at surface bellcrank

Rudder - right hand side of horizontal stabilizer at the rudder bellcrank

Sideslip synchro - in vane

The indicated airspeed transducer is located in the nose section adjacent to the inverter and is a Statham P69 pressure pickup with a range of ±1 psi. The normal accelerometer is a Statham A8 with a range of ±3 g and is located on the floor of the main cabin immediately forward of the hydraulic reservoir.

Excitation for the position and sideslip synchros is furnished by a 26 volt 400 cycle transformer located in the power and rudder chassis. The outputs of the nulling synchros in the position and sideslip circuits are rectified, filtered, and recorded directly. To record the outputs of the indicated airspeed and accelerometer, additional amplification is necessary and is provided by a Kay Lab 106A amplifier in each circuit. The recording circuits are very similar, each comprising a 400 cycle chopper, a step attenuator, a recording filter with a nominal corner frequency of three cycles/second (Figure 38), and Heiland type G-150 recording galvenometer.

The rolling velocity, yawing velocity and rolling acceleration recording circuits are located on the aileron surface servo chassis. Yawing angular acceleration is on the rudder stick and surface servo chassis, with differentiated sideslip on the power and rudder chassis and differentiated airspeed on the elevator surface servo chassis.

The artificial control signals that are to be recorded are picked off at some convenient place in the control signal amplifier circuits. Isolation of these circuits is provided by a cathode follower. Recording attenuators in all circuits on the recording control chassis are between the chopper and low pass filter. In other circuits where an isolating cathode follower is used step attenuators are provided in the grid circuit of the isolating tube.

CALIBRATIONS AND COMPONENT PERFORMANCE

SERVO PERFORMANCE

Transient responses to step inputs were obtained for all servos by unbalancing the feedback circuit with a switched-in resistance circuit. The input step and the servo strut response were recorded on the oscillograph. The transient responses were harmonically analyzed on a Dent-Draper Rolling Sphere Harmonic Analyzer. Feedback gains were those at which the servos are now set and were selected as discussed in the section on Artificial Force Feel.

The frequency response of the elevator force servo is presented in Figure 40. Transients of stick travel forward and aft for two different values of strut travel were analyzed. Considerable difference in amplitude response for stick forward and aft at the smaller strut travel of 3/8 in. is apparent; much less difference in response between the two directions exists for a strut travel of 3/4 in. This different response for direction of strut travel is not noticeable in operating the airplane, and is more pronounced with this servo than any other. This Moog transfer valve may be somewhat more non-linear for low flow values than other valves used. With the assumption of a second-order system, this servo has an undamped natural frequency of 4.5 cycles/second, with a damping ratio of approximately .77.

The aileron force servo frequency response as presented in Figure 41 indicates a somewhat higher natural frequency than the elevator force servo with a feedback gain control setting of only 30 as compared with 70 for the elevator force servo. The Moog valve for this servo is a more recently manufactured valve of the same model as the others. Internal valve ports are cut somewhat more accurately, and the flow curve of this transfer valve is considerably sharper through zero than the older models. This servo has an undamped natural frequency of 5.2 cycles/second with a damping ratio of approximately .70.

The rudder force servo response as presented in Figure 42 has an undamped natural frequency of 3.2 cycles/second with a damping ratio of approximately .63.

The elevator and aileron surface servo frequency responses are presented in Figures 43 and 44 respectively. Both these servos with feedback gain control settings of 100 have much higher natural frequencies than the force servos.

Analysis of the resulting faster transient responses was made rather inaccurate with the relatively slow paper speed available on the Heiland oscillograph. Therefore, the quantitative usefulness of these frequency responses is not too great.

The rudder surface servo frequency response is presented in Figure 45. The relatively low natural frequency of this servo with its feedback gain control setting of 100 is attributed to its low feedback gain: approximately one-half that of the other servos for this same gain control setting, as its servo amplifier has a low gain 12AU7 tube instead of a 12AT7 tube. The response of this servo may be improved to a value similar to that of the aileron and elevator surface servos by replacing the present tube with one of higher gain as discussed previously.

ARTIFICIAL FORCE FEEL

With control of the airplane through the hydraulic servos the pilot's controls are mechanically separated from the control surfaces while the co-pilot's remain connected. Force feel is provided through hydraulic servos directly connected to the pilot's controls. Pilot applied force is sensed by strain gages on the special control wheel for elevator and aileron controls and on the rudder pedal force attachment for rudder control. The output of the strain gage bridge is amplified, divided by dynamic pressure, and then fed to the servo mixer amplifier. Thus, the applied force orders a control position through the force servo.

The inherent inertia load outboard of the control force strain gages is relatively small for each of the three controls. However, control force application by the pilot markedly increases this inertia. This inertia may be considered an additional feedback loop that is destabilizing. Because of this inertia feedback there is a maximum permissible force input gain for particular feedback gains. Very rapid large amplitude control force application with greater force input gains initiates an oscillatory instability. However, removal of the pilot applied force stops the oscillation by reducing the inertia.

Force servo feedback gains have been set as high as possible to still allow attainment of the necessary force gradients with force input gains that are well below those for instability. The resulting natural frequencies and damping ratios

of the force servos have proven satisfactory. If, for any reason, greater values of force servo feedback and force input gains are desired, the force servos should be checked for instability at the dynamic pressure corresponding to the airspeed at which these values will be used. The values indicated below are satisfactory at zero airspeed which provides the highest force input gain through the present "q" divider, The values of gain are those on the respective servo input potentiometers on the control panel as presented in Figure 30:

CONTROL	FORCE INPUT GAIN	FORCE FEEDBACK GAIN	CALIBRATION @ 0 IAS LB. CONTROL FORCE PER DEGREE SURFACE TRAVEL
Elevator	40	70	3.3
Aileron	30-*65	30	1.7 - .8
Rudder	70-*90	30	5.9 - 4.6

* *The higher values of gain indicated for aileron and rudder have been checked in flight at airspeeds as low as 100 MPH.*

Gain controls for the force inputs and force and surface servo feedback circuits are on the test engineer's control panel as presented in Figure 29. The helipots controlling these gains are labelled as follows:

HELIPOT	CONTROL ITEM
G	Elevator force input
H	Aileron force input
J	Rudder force input
K	Elevator force servo feedback
L	Aileron force servo feedback
M	Rudder force servo feedback
N	Elevator surface servo feedback
O	Aileron surface servo feedback
P	Rudder surface servo feedback

STICK TO SURFACE GEARING

The gearing between the stick and surface servos is adjustable with potentiometers located on the respective surface servo instrumentation chassis. These potentiometers adjust the gain of the signal from the force servo feedback potentiometer which is then fed to the mixer of the surface servo. These gains have been set to provide a one-to-one ratio between the cockpit controls and surfaces; that is, the co-pilot controls move the same distance as the pilot controls with operation through the servos. The surface servo feedback gains are all set at 100. If these are changed, it is necessary to change the potentiometer settings to maintain a one-to-one ratio.

ARTIFICIAL STABILITY INPUTS

The amount of each artificial stability derivative provided is controlled by the respective potentiometer on the test engineer's servo input control panel. The maximum available value of each artificial derivative is presented in Table IV. The control surface travel per unit input is given for full potentiometer gains of 100 with the present dynamic pressure division for 120 MPH. The approximate equivalent incremental derivatives are given for 120 MPH at 5000 feet with a gross weight of 8200 lbs., and a center-of-gravity position of 28% MAC. These artificial derivative values must be considered approximate, as calculated control power derivatives were used in determining them. Also, the values of derivatives given are absolute, with no correction for phase lag between airplane response input and surface.

The inherent phase lags of the artificial input components at representative frequencies as calibrated are listed in the table on the following page.

PHASE LAG - DEGREES

INPUT	FREQUENCY -CYCLES/SEC.	SENSING ELEMENT	CONTROL FILTER	SURFACE SERVO	TOTAL SURFACE/INPUT
$C_{m_{DV}}$	.033	1	1	1	3 (Approx.)
C_{ℓ_r}	.52	3	66	4	73 (Filter #4)
$C_{\ell_{Dp}}$	.52	7	15	4	26
C_{n_β}	.31	-	7	6	13
C_{n_r}	.55	3	16	10	29
$C_{n_{D\beta}}$	.39	5	10	7	22
$C_{n_{Dr}}$	.55	7	16	10	33

The maximum available values of the artificial stability derivatives as presented in Table IV are not necessarily obtainable as linear functions for all possible airplane maneuvers. Flight tests have indicated no need for the maximum values as available in order to provide a damped airplane. If, for any reason these values are desired, it should be noted that the control deflections available will not necessarily provide the artificial derivative indicated for large amplitude maneuvers. For example, an abrupt rudder deflection of approximately 13 degrees at 100 MPH with the maximum available artificial C_{ℓ_r} would result in full aileron travel to provide this artificial stability.

RECORDING

The oscillograph switch may be operated at the oscillograph itself or from the spring switch on the test engineer's control panel. The items recorded, their trace identifications, and calibrations are presented in Table V.

DYNAMIC PRESSURE DIVISION

Division of artificial control forces and some of the artificial derivatives, as indicated in Table IV, was quite satisfactory with the original Giannini pressure transmitter for flights 5-16 as indicated in Figure 46. Replacement of this transmitter due to faulty operation with one of the same type resulted in dynamic pressure division that is much less satisfactory at airspeeds below 150 MPH as indicated in Figure 46. The transmitter presently installed has a very non-linear output with pressure for the range 0 - .3 psi. Replacement of this transmitter with one of a more suitable type with an operating range of 0 - .7 psi instead of the present 0 - 4 psi will greatly improve the dynamic pressure division.

OPERATION OF ARTIFICIAL STABILITY SYSTEM

Approximately 300 hours of ground operation of the hydraulic system has been accomplished in the development, calibration, and operational check-out of the artificial stability system. The electronic equipment has been operated approximately three times as much. Nineteen flights with a total of 41½ hours have been made in the flight test phase to determine the artificial stability required and to demonstrate the system. General operating procedures for the artificial stability system including detailed engaging procedures are presented in this section.

GROUND OPERATION

For ground operation of the hydraulic system, quick disconnects are provided in the pressure and return lines in the starboard engine nacelle. A gear pump driven by an electric motor has been used as a ground pressure supply with cooling provided by a water-cooled heat exchanger.

A procedure for checking operation of the artificial stability system has been developed. This procedure may also be used as a preflight check out. Figure 30 presents a schematic of the instrumentation chassis with the adjustment potentiometers and test jacks. Test jacks are labelled on each chassis. Figure 29 presents a photograph of the test engineer's servo input and hydraulic control panels.

Ground operation requires a suitable rectifier connected into the airplane's power input jack and a hydraulic mule connected into the hydraulic system at the right engine nacelle with a heat exchanger connected into the hydraulic return line. The operational check out procedure is as follows:

OPERATIONAL CHECK OUT PROCEDURE FOR ARTIFICIAL STABILITY INSTALLATION

1. Hydraulic manual valve open (system pressure zero); pilot's by-pass switches OFF; control separation handles forward; test engineer's by-pass and pressure switches OFF and servo switches in TUNE; power switch OFF.

2. Check DC balance of all Moog servo valves. With an ohmmeter connected in

the pertinent test jacks and using the DC balance potentiometer for each servo valve, equalize valve coil resistances. The resistance of only one coil of the valve can be varied with the balancing potentiometer. The DC balance varies between valves and is in the range 800-1200 ohms.

3. Power switch ON. Push button to reset power relay on aft instrumentation chassis. Servo signal lights in test engineer's control panel should be on. External power should be such as to give a system frequency of 390-410 cycles/second. An AC outlet is available at the rear of the aft instrumtation chassis for a frequency meter and an AC vacuum tube voltmeter.

4. Balance the following with their respective balancing potentiometers and check with an AC vacuum tube voltmeter connected in the pertinent test jacks

SERVO INPUTS	APPROX. BALANCE - VOLTS
*a. Aileron force bridge } aileron and elevator stick control chassis	.02
*b. Elevator force bridge } aileron and elevator stick control chassis	.02
*c. Rudder force bridge - Rudder chassis	.01
d. Sideslip (vane locked in neutral) Power and rudder chassis	.03
e. Differentiated sideslip - Power and rudder chassis	.03
f. Yaw acceleration - Rudder chassis	.02
g. Roll acceleration - Aileron surface chassis	.02

** Phase and resistance balance potentiometers are , ovided for use in combination to balance force bridges.*

RECORDING	
a. Aileron position } Recording chassis	.01
b. Elevator position } Recording chassis	.01
c. Rudder position } Recording chassis	.01

RECORDING (cont'd.)

d. Normal acceleration	Recording chassis	.02
e. Sideslip	Recording chassis	.02
F. Airspeed dynamic pressure	Recording chassis	.05

5. With respective position balance potentiometer balance each of the six servos so that voltages across each side of Moog valve are equal. Connect DC meter on 10V scale in transfer valve test jacks on respective servo chassis.

6. Set servo input gain settings to proper values.

7. Test engineer's servo, pressure and by-pass switches ON.

8. Push elevator and rudder disaster detector reset switches. This insures the relays being reset in the event operation of servo switches caused disaster detector relay operation.

9. Close hydraulic manual valve bringing system pressure up to 1500 psi.

10. Pilot's separation control handles aft (singly or in combination).

11. Pilot's by-pass switches ON (singly or in combination).

12. Cycle all six servos through full travel with position balance potentiometers to check operation and remove any trapped air.

    Note: Servo struts reach full travel before their respective position balance potentiometers. If potentiometers are moved considerably beyond this point in cycling the elevator and rudder surface servo struts, the disaster detectors will operate.

13. Check force gradients over speed range. Set fixed pressure at airspeed pitot to simulate different airspeeds. Move pilot's controls and check force feel for linearity and absence of dead spots. Check force gradients for proper magnitude and variation with airspeed indicating proper q divider operation.

14. Check pressure valve operation. Move the separation control handles forward thereby closing the pressure valves and cutting off current to the servo Moog valves. This operation removes the servos from the control system if the pressure valves are operating properly even with the by-pass switches ON. Cycle cockpit controls manually for freedom.

15. Check by-pass valve operation. With control separation handles aft so pressure valves are open and with by-pass valve switches OFF, remove electrical connector from Moog transfer valve. Servos should be effectively removed from control system if by-pass valves are operating properly. Cycle cockpit controls manually for freedom.

16. With servos normally engaged check emergency disengage switches on pilot and copilot control wheels. Operation of either of these switches cuts off current to the pressure valves, by-pass valves, and servo amplifiers. Power reset switch must be reset after each usage of emergency disengage switch before servos can be re-engaged. Before resetting power relay, control separation handles should be forward and by-pass switches OFF. Re-engage servos normally to insure against excessive jump in controls when servos are re-engaged.

17. Check elevator and rudder disaster detector operation. Connect a DC meter to the E valve test jack in the disaster detector chassis which measures system excitation voltage (28 volts) at the output of the disaster detector. With hydraulic pressure manual valve open (system pressure zero) and current to servo amplifiers ON, unbalance respective surface servo input with position balance potentiometer until the disaster detector output voltage drops to zero. Read the voltage across the disaster detector from ground to the test jacks labelled EG1 and EG2 on the disaster detector chassis. The relays are set to operate at approximately 8-9 volts in either direction. The voltages at which the relays will operate is adjusted with the potentiometers above the EG1 and EG2 test jacks.

FLIGHT OPERATION

The test engineer at his station in the cabin balances the servo inputs and

recorded items, sets the desired servo input gains, closes the hydraulic system "short circuit" valve, and places his servo and hydraulic system control switches in the ON position. Only after the test engineer's control switches are on, can the pilot engage the servos. The pilot then moves the separation control handles to disengage his normal controls and places the pilot's by-pass switches in the ON position. The airplane is then controllable by the pilot through the servo system.

Balance checks of the servo inputs are accomplished with the airplane in trimmed level flight. Force bridge balancing must be done with the pilot's hands and feet off the controls. Normally, these bridge balances hold and do not have to be rebalanced in flight. Null balancing of the sideslip follow-up synchro is necessary in the trimmed flight condition to insure against an out of trim signal to the rudder when the sideslip to rudder input gain is set. Null balancing of the differential sideslip follow-up synchro is necessary only to maintain linear operation of this synchro. As the synchro in the sideslip vane is geared one-to-one with the vane, and autosyns are normally linear for at least ±12 degrees, an unbalance of several degrees would be allowable for normal values of sideslip experienced in flight.

Position balancing of each servo so that the voltages across each side of the valve are equal, is the primary function of the test engineer before engaging the servos. If this is accomplished with reasonable accuracy, the change in position of the pilot's cockpit controls and the control surfaces when the servo system is engaged will be negligible.

Pilot trimming of the airplane controls with changes in airspeed or airplane configuration while operating through the servos is accomplished through the normal trim tab wheels. The helipots geared to these tab wheels position the force and surface servos. The gains of the trim helipot outputs have been adjusted so the surfaces move approximately the same amount for a given tab travel with operation through the servos as with the normal airplane.

A step-by-step procedure for engaging the artificial stability system has been developed and is as follows: (See next page.)

ENGAGING PROCEDURE FOR ARTIFICIAL STABILITY EQUIPMENT

1. With hydraulic system pressure zero, pilot's by-pass switches OFF, control separation handles forward with pilot engaged to normal controls, test engineer's servo switches in TUNE and pressure and by-pass switches OFF, power switch ON.

2. Reset power relay on aft instrumentation chassis.

3. With AC vacuum tube voltmeter check balance of following:

ITEM SERVO INPUTS	APPROX. BALANCE - VOLTS
*a. Aileron force bridge	.05
*b. Elevator force bridge	.05
*c. Rudder force bridge	.05
d. Sideslip	.07
e. Differentiated sideslip	.08

* *These items normally hold balance and do not requirè rebalancing in flight.*

RECORDING

a. Airspeed dynamic pressure.

b. Sideslip.

c. Elevator position - Will need balancing only if attenuation 1 is used.

d. Normal acceleration.

4. With DC meter position balance all six servos so that voltages across each side of Moog valve are equal. Use transfer valve test jacks.

5. Set servo input gains on test engineer's control panel to provide desired

force gradients and artificial stability.

6. Test engineer's servo, by-pass, and pressure switches ON.

7. Push elevator and rudder disaster detector reset switches.

8. Close hydraulic manual valve bringing system pressure up to 1500 psi.

9. Pilot move separation control handle aft.

10. Pilot's by-pass switch ON.
    Airplane now controllable through servos.

Each system singly. Check each control system for proper operation before engaging next system.

11. In the event the elevator or rudder disaster detector operates to remove the respective servo system from operation, pilot re-engages normal control for that control system. With by-pass and pressure switches OFF and servo in tune determine the source of the signal that operated the detector relay. Re-engage with normal procedure when malfunction is remedied.

12. Normal servo disengage and control re-engage procedure:

    (a) Pilot's by-pass switches OFF

    (b) Separation control handles forward

13. Emergency servo disengage: Push emergency button on either pilot or co-pilot control wheels or test engineer's power switch OFF.

TROUBLE SHOOTING

Sufficient operating time has been obtained with the artificial stability system to experience some malfunctions of the hydraulic and electronic components. A number of these items are presented in Table VI with their associated symptoms and possible fixes.

The symptom of sharp small pulses in the co-pilot's control which indicates extraneous signal to the surface servo may be the result of a noisy feedback in

either the force or the surface servo. It can result from the force servo feedback and not be noticeable in the pilot's controls. The surface servos in each case have higher feedback gains than the force servos as discussed previously in the section on artificial force feel. This higher feedback necessitates a larger gain in the signal from the force servo feedback potentiometer to the surface servo than to the force servo itself. Thus, small amplitude noise can be transmitted to the surface servo without being large enough to be noticed in the force servo. Normally, cleaning the faulty potentiometer in carbon tetrachloride and applying a light potentiometer grease has eliminated the difficulty for several flights.

The difficulty with noisy feedback potentiometers has occurred in both the aileron and elevator control systems, although only once in the elevator system. During the flight phase, it was necessary to clean or replace the aileron force servo feedback potentiometer twice and the aileron surface servo feedback potentiometer once. The aileron servo installations probably collect more dirt than the others and it may prove desirable to cover this installation to reduce the possibility of dirt and hydraulic oil entering the potentiometers.

The difficulty with noisy tubes has occurred only twice since the beginning of the flight phase. Care has been taken to provide low noise tubes, particularly in the mixer amplifiers.

Considerable difficulty was experienced with the modified pressure valves during the ground check-out phase. Also, two malfunctions of these valves were experienced during the flight phase: one failed to close, and one failed to open. With this valve it is difficult to set the spring preload high enough to insure proper valve closing when the solenoid is de-energized and still insure sufficient solenoid force to open the valve properly. No difficulties have been experienced with the Saval pressure valves which are now installed for the two elevator servos.

An open circuit developed in the Giannini pressure transmitter on two occasions (q divider). Replacement of this transmitter with a more suitable type has been recommended previously in the section on dynamic pressure division.

No difficulties have been experienced in this installation with dirt-clogged

Moog transfer valves. With proper precautions in adding fluid to the hydraulic system and changing filter elements, troubles of this nature will be minimized.

Slow leaks due to "O" ring wear in the servo strut piston seals and transfer valve seals have occurred in three servo installations over the entire operating period. This condition is easily corrected and does not appear to be abnormal.

Due to manufacturing tolerances in the splines of the control separations, some play exists in the normal airplane controls between the pilot's controls and the control surfaces. The aileron control separation uses four welded keys as splines. After several flights the play in this control became considerably greater. The separation shaft was removed and chrome plated. The play now in all controls is not objectionable.

HANDLING CHARACTERISTICS WITH ARTIFICIAL STABILITY

No extensive pilot evaluation program of the artificially stabilized airplane was accomplished as this was not the purpose of this project. After the flight shakedown of the equipment was completed and a brief calibration made of the effect of the artificial stability on the handling characteristics, C.A.L. pilots made qualitative pilot evaluation checks of various artificially stabilized configurations. On two flights fairly rough air conditions were encountered and qualitative data was obtained for various configurations. Maj. G. Lairmore and Mr. D. Graham of the All Weather Flying Section, Flight Test Division, Wright-Patterson Air Base also flew the artificially stabilized airplane as did Capt. F. E. Davis, Chief Engineering Pilot for the eastern division of Eastern Air Lines.

In this section of the report the ability of the artificial stability installation to accomplish the desired damping and handling characteristics will be discussed. Also, a brief summary of pilot opinion will be presented.

PHUGOID DAMPING

Damping of the phugoid mode of longitudinal motion is accomplished quite well, particularly at approach airspeeds with the addition of $C_{m_{\dot V}}$ or rate-of-change of airspeed to the elevator. A reproduction of oscillograph records of the phugoid oscillation at 120 MPH trim speed of the normal airplane with $C_{m_{\dot V}} = 0$ and with an approximate value of $C_{m_{\dot V}} = 1.5$ is presented in Figure 47. Values of derivatives must be considered approximate as they are based upon calculated control power derivatives, upon airplane gross weights as estimated from known take-off gross weights, and density altitudes estimated as equal to pressure altitudes.

With pull up to 108 MPH from 120 MPH trim airspeed and release with $C_{m_{\dot V}} = 1.5$ the airspeed returns to trim with no overshoot. However, there is a residual oscillation in airspeed about trim of approximately 2 MPH double amplitude. At 100 MPH an essentially dead-beat phugoid is possible with approximately 1 MPH double amplitude oscillation about trim. At 150 MPH the residual oscillation about trim is approximately 3-4 MPH double amplitude.

DUTCH ROLL DAMPING

Dutch roll damping may be accomplished with either C_{n_r} or $C_{n_{D\beta}}$. As indicated in the theoretical analysis $C_{n_{Dr}}$ will also provide Dutch roll damping. However, flight test results indicated insufficient $C_{n_{Dr}}$ available in the present installation, and project time was too limited to allow further work on this channel. It is hoped that an investigation of the effect of $C_{n_{Dr}}$ can be conducted at some future time.

The effect of artificial $C_{n_{D\beta}}$ on the Dutch roll motion is presented in Figure 48. The Dutch roll oscillation was initiated with a rudder kick and release as shown in the normal airplane configuration. The continuing rudder motion after release in the configuration with $C_{n_{D\beta}} = .053$ at 120 MPH is in response to the rate-of-change of sideslip signal. This value of $C_{n_{D\beta}}$ provides Dutch roll damping that is somewhat greater than necessary. A value of $C_{n_{D\beta}} = .044$ provides esentially dead beat damping.

Dutch roll damping is also accomplished with artificial C_{n_r} as is indicated in Figure 48. This rudder kick is in the opposite direction from the other responses in Figure 48, and the sense of the rudder trace is reversed. This value of $\Delta C_{n_r} = -.42$ at 120 MPH provides essentially dead beat damping of the Dutch roll.

Although artificial C_{l_r} is provided for spiral stability, its effect on Dutch roll damping is of interest. In Figure 49 is presented a rudder-initiated Dutch roll oscillation with $\Delta C_{l_r} = -.30$ at 120 MPH. A slight decrease in damping over that of the normal airplane as well as a slightly longer period is indicated. With filter #4 as employed in this transient response, the roll-to-yaw ratio is reduced from 1.6 for the normal airplane to 1.0 for $C_{l_r} = -.30$. With an available gain 10 times that used at this airspeed, corresponding increases in the effect of this artificial derivative on the lateral motion can be realized.

The effect of C_{n_β} on the Dutch roll oscillation is presented in Figure 49 for an approximate value of $\Delta C_{n_\beta} = .05$ at 120 MPH. When this rudder-initiated transient is compared with that for the normal airplane, a decrease in damping and a shorter period of the Dutch roll are apparent. The decrease in damping

is primarily the result of the destabilizing $C_{n_{D\beta}}$ added due to the inherent phase lag between the rudder and the sideslip input. If artificial C_{n_β} is desired, Dutch roll damping can be maintained by adding greater values of $C_{n_{D\beta}}$ to compensate for the destabilizing effect of the lag in C_{n_β}. A divergent Dutch roll is attainable with the addition of ΔC_{n_β} of approximately .10.

TURN COORDINATION

With $C_{n_{D\beta}}$ for deadbeat Dutch roll damping, turn coordination is maintained over the speed range within one-third ball width for abrupt entry into needle width turns using ailerons only. The steady state coordination in a needle width turn is less than one-eighth ball width.

Turn coordination with C_{n_r} for dead beat Dutch roll damping is maintained within one-quarter ball width for abrupt entry into a needle width turn with a steady state coordination of one-eighth ball width.

DAMPING IN ROLL

The addition of artificial $C_{l_{Dp}}$ increases the damping in roll by decreasing the effective inertia in roll. Configurations have been flown with values of $C_{l_{Dp}}$ as high as .034 at 120 MPH, which is an equivalent reduction in roll inertia of approximately 65%. This reduction of inertia results in a more snappy response in roll that is quite noticeable in the handling characteristics. No analysis of the flight test data available has been made to determine the actual increase in damping of the roll root of the lateral stability characteristic equation.

SPIRAL STABILITY

Spiral stability is provided both through C_{n_r} and C_{l_r}. The amount available through C_{n_r} is limited as the Dutch roll damping is very great before large increases in spiral stability are realized.

The normal airplane is very slightly spirally divergent in flight. Values of artificial ΔC_{ℓ_r} as small as .10 are sufficient to provide slight spiral convergence.

PILOT OPINION

It must be emphasized that only very brief pilot evaluation tests were conducted; so that the opinions reported are more in the nature of first impressions. A simple investigation procedure was used to point up the more obvious differences in handling characteristics and was as follows:

1. From steady state sideslip of approximately 5 degrees, release rudder abruptly.

2. Perform relatively steep turn and recover.

3. Perform gentle turn and recover.

4. Fly hands off for 1-2 minutes.

The addition of phugoid damping was of course quite obvious when the airplane was disturbed from trim with the elevator and then released. However, in normal flying this additional damping did not affect the pilot opinions on handling characteristics, particularly professional pilots. With pilots of less experience, the additional phugoid damping was immediately noticeable in improving the ability to maintain constant airspeed during turns. In straight and level flight through moderately rough air there was no noticeable difference in handling characteristics with or without artificial phugoid damping. It is felt that instrument let-downs and approaches would provide a good evaluation of the effect of additional phugoid damping.

Additional Dutch roll damping as provided with C_{n_r} provided a better ride in rough air than with $C_{n_{DA}}$ and considerably better than the normal airplane. However, with sufficient C_{n_r} to provide an essentially dead beat Dutch roll, rudder pedal forces in turns were objectionably high with the pedal force gains set for the normal airplane. When the pedal force gradient was reduced by increasing the force input gain to the force servo this objection to the high value of C_{n_r} was removed.

With artificial $C_{n_{D\beta}}$ to provide dead beat Dutch roll damping the rudder forces in turns were not objectionable. The improved turn coordination with $C_{n_{D\beta}}$ over C_{n_r} requires less rudder for coordination. In fact, with $C_{n_{D\beta}}$ little coordination is necessary. In general, the advantage of additional Dutch roll damping was observed primarily in rough air.

The more snappy response of the airplane in roll with artificial $C_{\ell_{Dp}}$ added to decrease the effective inertia provides a precise feel to the controls that was felt to be a definite improvement. No check with high values of this input was made in rough air.

The additional spiral stability with artificial C_{ℓ_r} was immediately apparent and considered desirable, particularly in instrument flying. C_{ℓ_r} added in an amount large enough to provide "automobile" type aileron control, in which a given wheel position orders a given bank angle, was not liked by the professional pilots. The necessity of holding ailerons on in a turn even with reduced aileron force gradients was objectionable. To less experienced pilots this feature had some merit. A more thorough evaluation of this type of aileron control should prove enlightening.

The following configurations which provide heavily damped Dutch roll, spiral and phugoid modes were in general considered to quite satisfactory and a considerable improvement over the normal airplane:

1. $C_{m_{DV}} = 1.5$

   $\Delta C_{n_r} = -.21$

   $\Delta C_{\ell_r} = -.30$, filter #4

GAIN SETTINGS

CONTROL	FORCE INPUT	FORCE FEEDBACK	SURFACE SERVO FEEDBACK
Elevator	30	70	100
Aileron	25	30	100
Rudder	85	30	100

2. $C_{m_{DV}} = 1.5$

$C_{n_{D\beta}} = .044$

$\Delta C_{\ell_r} = -.30$, filter #4

Rudder force input gain = 70, other force and surface gains same as (1). An additional configuration that provides a heavily damped airplane with somewhat faster roll response and provides "automobile" type aileron controls is as follows:

$C_{m_{DV}} = 1.5$

$C_{n_{D\beta}} = .044$

$\Delta C_{\ell_r} = -.90$, filter #4

$C_{\ell_{Dp}} = .035$

Force and surface gains same as (2).

CONCLUSIONS

Artificial stabilization of a C-45 airplane has been accomplished resulting in essentially dead beat phugoid and Dutch roll motions. Damping in roll has been increased, airplane response in roll has been made more precise; and stability of the spiral motion can be increased to any desired degree. The components necessary to provide a more precise airplane response to rudder motion have been incorporated in the control system. However, this feature has not yet been checked in flight.

The inherent flexibility of the artificial control system as installed in the C-45 airplane provides considerable potentiality for investigation of airplane handling characteristics. Elevator, aileron and rudder force gradients can be varied over a wide range as can each of the seven artificially added derivatives presently available: $C_{m_{DV}}$, C_{n_r}, $C_{n_{D\beta}}$, C_{ℓ_r}, $C_{\ell_{Dp}}$, $C_{n_{Dr}}$ and C_{n_β}. Effective hinge moment characteristics of the surface controls can be varied by adjusting the gear ratios between the cockpit and surface controls. With minor circuit modifications, the additional artificially added derivatives C_{ℓ_p}, C_{n_p} and C_{ℓ_β} would be available. Reorientation of one of the existing rate gyros and one of the amplifier circuits could provide artificially added $C_{m_{D\theta}}$ if modification of the longitudinal short period motion is desired. Also, the derivative C_{m_V} may be added with circuits that exist in the installation.

With these added derivatives available, the possible variations in period and damping of the airplane's motions, roll to yaw ratio, and static control characteristics are almost limitless within the capabilities of the airplane's control surfaces.

The limited pilot evaluation obtained by the artificially stabilized airplane indicated the need for a thorough investigation to determine those handling characteristics which best permit the pilot to perform the mission of a particular type of aircraft. This installation in the C-45 airplane is particularly suited to such an investigation.

REFERENCES

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2. Notess, Charles B. and Kidd, Edwin A. *Analog Computer Analysis of Artificial Methods for Stabilizing the Lateral Modes of a C-45 Airplane.* C.A.L. FRM No. 150, 1 April 1953.

3. Kidd, Edwin A. *Mechanical Installations in C-45, All Weather Flying.* C.A.L. FRM No. 168, 19 January 1953.

4. Moog, William C., Jr. *Direct Current Nozzle Drive Transfer Valve.* C.A.L. Report No. ID-389-S-4.

TABLE I

PERTINENT DIMENSIONS AND DETAILS OF THE C-45 AIRCRAFT

GENERAL DIMENSIONS

WINGS	
Span	47.65 ft.
Airfoil Section	
Chord at root	NACA 23020
Chord at tip	NACA 23012
Root Chord (Theoretical at ℄ of fuselage)	135.12 in.
Tip Chord (Theoretical at outer end of tip)	42 in.
Incidence	
Root	3.92°
Tip	.38°
Dihedral (at 25% of the chord aft of leading edge)	6°
Sweepback (at 25% of the chord aft of leading edge)	7.08°
STABILIZER	
Span	13.45 ft.
Maximum Chord	5.83 ft.
FIN	
Span	5.42 ft.
Maximum Chord	4.02 ft.

AREAS

Wings (total)	349 sq. ft.
Ailerons (total)	22.84 sq. ft.
Stabilizer (including elevator)	65.40 sq. ft.
Elevator (including tabs)	29.44 sq. ft.
Fins (total)	16.30 sq. ft.
Rudders (including tabs)	17.28 sq. ft.

RANGES OF MOVEMENT OF CONTROL SURFACES

Aileron	- Up	45 deg.
	Down	20 deg.
Elevator	- Up	35 deg.
	Down	25 deg.
Rudder	- Right	25 deg.
	Left	25 deg.
Trim Tabs:		
Elevator	- Up	20 deg.
	Down	14 deg.
Rudder	- Right	30 deg.
	Left	30 deg.
Aileron	- Up	20 deg.
	Down	20 deg.

TABLE II

INSTALLATION DRAWINGS

DRAWING NUMBER	TITLE
FRS-359-004-3	Elevator Separation Components
FRS-359-004-4	Elevator Separation Support
FRS-359-004-5	Rudder Separation Assembly
FRS-359-004-6	Aileron Control Modification and Servo Installations
FRS-359-004-7	Rudder Force Servo Installation
FRS-359-004-8	Control Pedestal Assembly
FRS-359-004-9	C-45 Hydraulic System Schematic
FRS-359-004-10	Elevator Position Servo and Bellcrank Installation
FRS-359-004-12	Rudder Servo and Bellcrank Installation in Horizontal Stabilizer
FRS-359-004-13	Elevator Force Servo Installation
SH-2032	Piston - Flight Research Servo
SH-2034	Housing - Flight Research Servo
SH-2038	Assembly - Flight Research Servo
SH-2043	Body - By-pass Valve
SH-2044	Piston Seal - By-pass Valve
SH-2045	Coupling - By-pass Valve
SH-2046	Bushing Rework - By-pass Valve
SH-2047	Spool Rework - By-pass Valve

PROJECT SKETCH NUMBER	
1058	Rudder Surface Servo Bellcrank
1060	Rudder Surface Servo Shaft Extension
1064	Servo Strut Attachment
1067	Elevator Surface Control Bellcrank
1075	Rudder Pedal Servo Attachment
1103	Aileron Control Modification Pulley Bracket #2
1105	Aileron Control Modification Pulley Bracket #1
1106	Aileron Disengage Yoke Hinge Fitting
1120	Actuator Brackets (Rudder Separation)

TABLE II (cont'd.)

PROJECT SKETCH NUMBER	TITLE
1123	Elevator Disengage Clevis Ends
1125	Aileron Disengage Pulley Brackets
1126	Rudder Disengage Pulley Bracket #1
1127	Rudder Disengage Pulley Bracket #2
1129	Aileron and Rudder Disengage Handle
1130	Control Pedestal Clevis Ends
1131	Elevator Disengage Handle
1132	Control Pedestal Clevis End
1133	Aileron Actuator Bellcrank
1136	Elevator Surface Servo Rod End
1137	Elevator Force Servo Rod End
1138	Rudder Separation Spring End Attachments
1140	Aileron Separation Spring Guide
1141	Control Column Servo Attachment
1459	Rudder Separation Bellcrank
1461	Rudder Force Bellcrank
1462	Rudder Separation Shaft
1463	Female Spline with Male Dog
1472	Aileron Surface Control Bellcrank (Center)
1474	Aileron Surface Control Bellcrank (Inboard)
1476	Aileron Surface Control Bellcrank (Outboard)
1478	Aileron Surface Control Disengage Shaft and Collar
1489	Splined Collar Actuator

TABLE III

WIRING DIAGRAMS

DRAWING NUMBER	TITLE
359-003-1	Aileron Surface Servo Control
359-003-2	Rudder Pedal and Surface Servo Control
359-003-3	Hydraulic Control System
359-003-4	Servo Chassis Interconnecting Cabling
359-003-5	Recording Control Chassis
359-003-6	Test Engineer's Control Box Wiring
359-003-7	Power and Rudder Servo Controls
359-003-8	Chassis Wiring Divider Unit
359-003-9	Aileron and Elevator Stick Servo Control
359-003-10	Elevator Surface Servo Control
359-003-11	Elevator and Rudder Surface Control
359-003-12	Artificial Stability Sensing Pickup Wiring
359-003-13	Nose and Main Cabin Wiring

TABLE IV

ARTIFICIAL STABILITY INPUTS

IAS = 120 MPH
Altitude = 5000 ft.
Control Potentiometer Gains = 100

Gross Weight = 8200 lb.
C.G. Position = 28% MAC

INPUT	"q" DIVISION	HELIPOT	SURFACE TRAVEL PER UNIT INPUT	NORMAL AIRPLANE DERIVATIVE	INCREMENTAL ARTIF. DERIV. AVAILABLE
Differentiated airspeed to elevator - $C_{m_{DV}}$	yes	A	-3.9 deg./mph/sec.	0	4.0
Yaw velocity to aileron - C_{ℓ_r}	no	B	5.2 deg./deg./sec.	.032	-3.0
Roll acceleration to aileron - $C_{\ell_{Dp}}$	yes	C	-.24 deg./deg./sec.^{2}	0	.07
*Sideslip to rudder - C_{n_β}	no	D	-6.4 deg./deg.	.138	.50
*Yaw velocity to rudder - C_{n_r}	no	D	1.2 deg./deg./sec.	-.081	-.70
Differentiated sideslip to rudder - $C_{n_{D\beta}}$	yes	E	-2.3 deg./deg./sec.	0	.13
Yaw acceleration to rudder - $C_{n_{Dr}}$	yes	F	-.47 deg./deg./sec.^{2}	0	.13

* *These inputs are controlled with the same helipot. Selection of either is available with a switch in the aft instrumentation chassis.*

TABLE V

RECORDED CALIBRATIONS AND TRACE IDENTIFICATION

IDENTIFICATION	GALVANOMETER	ITEM	ATTENUATOR	CALIBRATION
1	1	Sideslip	2	6.6 Deg/In
2	2	Elevator position	1	1.8 Deg/In
			3	8.6 Deg/In
3	3	Right Aileron position	2	5.4 Deg/In
4		Dummy		
5	4	Rudder position	2	8.7 Deg/In
6	5	Normal acceleration	1	.8 g/In
7	*9	Airspeed - dynamic pressure	1	.123 psi/in
8		Dummy		
9		Dummy		
10	19	Roll velocity	9	
11	20	Yaw velocity	4	9.6 Deg/Sec/In
12	21	Roll acceleration	7	41.4 Deg/Sec^2/In
13	22	Yaw acceleration	5	27.9 Deg/Sec^2/In
14	23	Differentiated side-slip	2	5.0 Deg/Sec/In
15	24	Differentiated air-speed	3	1.6 MPH/Sec/In

** Galv. 9 is connected to galv. 6 input*

TABLE VI

GENERAL TROUBLE SHOOTING

	SYMPTOM	POSSIBLE MALFUNCTION	FIX
1.	Sharp, small amplitude pulses in copilot's controls, more or less continuous.	Noisy feedback potentiometer in either force or surface servo.	Switch to other potentiometer winding, clean potentiometer in carbon tetrachloride, or replace potentiometer.
2.	Occasional pulses in either force or surface servos.	Microphonic tube, most likely 6AU6 tube in mixer.	Replace tube
3.	After servos engaged, ability to move pilot's controls freely with no force feel. Also ability to move surface controls freely from copilot's side after servos engaged.	Pressure valve to respective servo not open.	Operate valve by hand several times. If still inoperative check solenoid.
4.	All pilot's controls essentially locked, when servos are engaged.	"q" divider malfunction - pressure pick-up inoperative.	Check Giannini pressure pick-up for open circuit. Repair or replace.
5.	Abrupt pulse in particular control upon dumping of hydraulic pressure with servos disengaged.	Pressure valve to respective servo not closed.	Repair or replace pressure valve. If modified 4-way type, increase preload on spring.
6.	Sluggish servo operation - slow response.	(a) Pressure valve to servo only partially open (b) Dirt clogged Moog valve (c) Open by-pass valve	(a) Same as (3) (b) Return Moog valve to Moog Valve Co. for cleaning. (c) Check by-pass valve for proper operation
7.	Hydraulic leak end of strut	Worn "O" ring on end plug	Replace "O" ring
8.	Hydraulic leak between Moog valve and strut	Worn "O" rings on base of Moog valve	Remove Moog valve and replace "O" rings
9.	Excessive play in normal airplane controls	Control separation spline wear	Remove respective splined shaft and chromeplate.

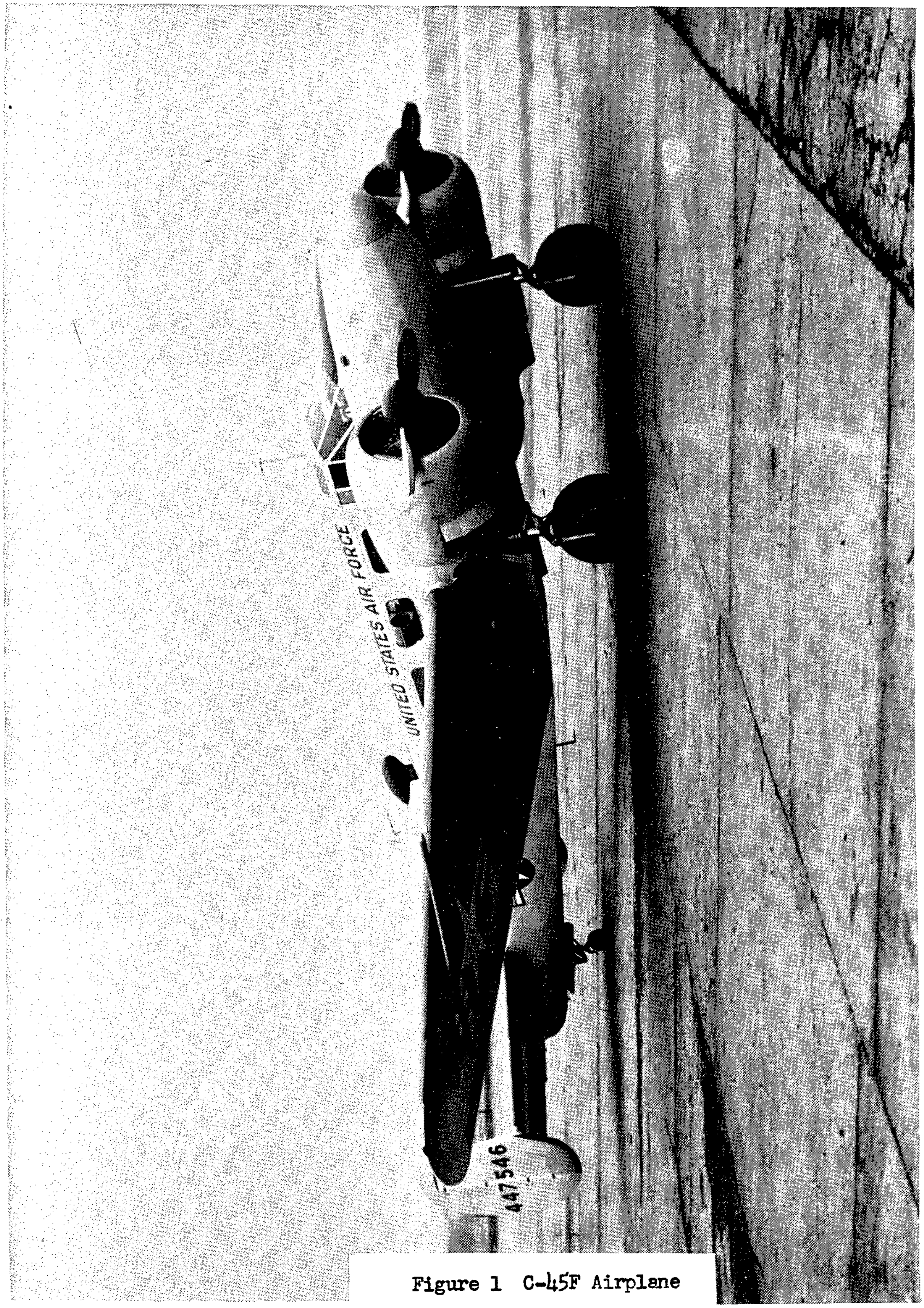
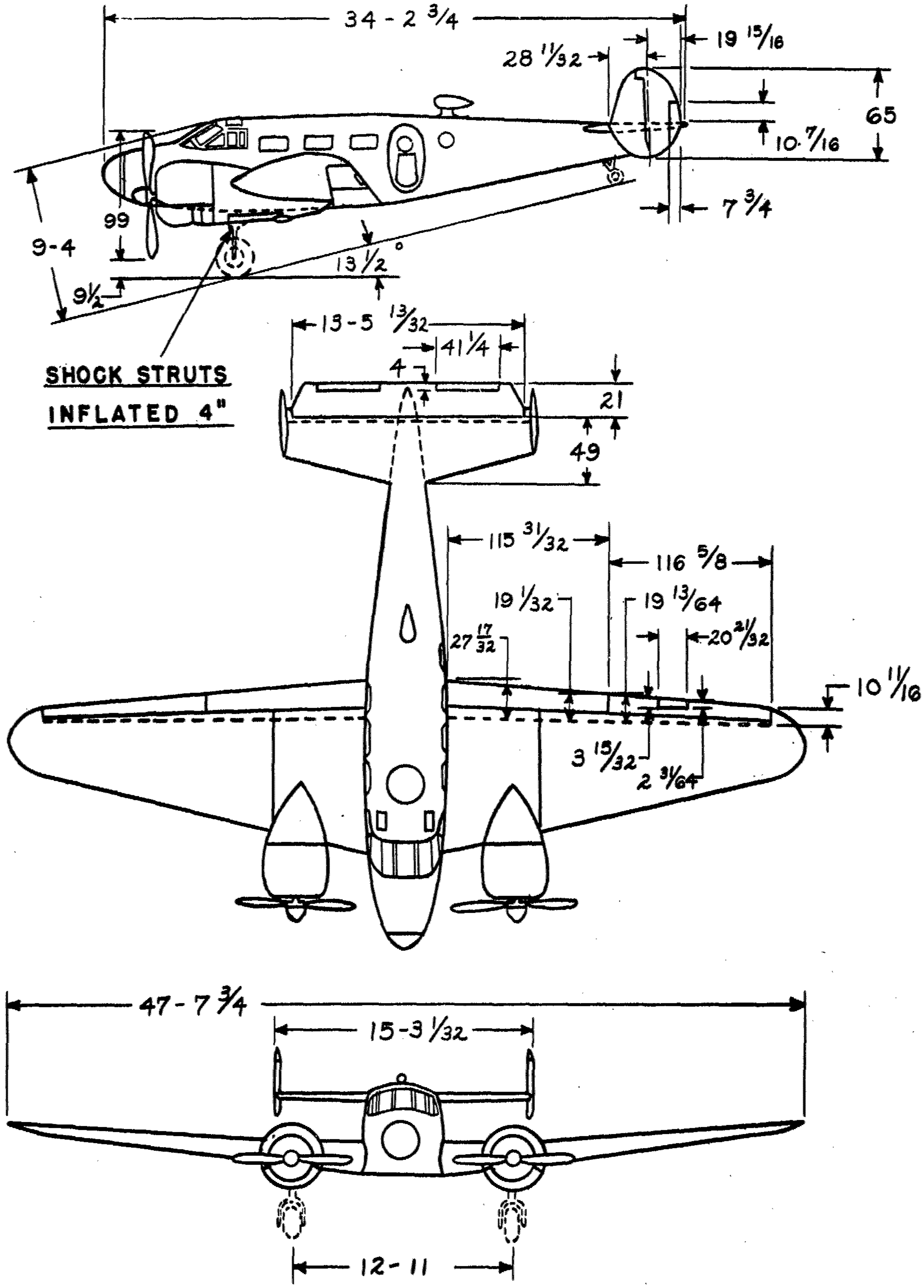


Figure 1 C-45F Airplane



C-45F THREE VIEW DIMENSIONAL DIAGRAM

FIG. 2

C-45F

EFFECT ON LONGITUDINAL STABILITY QUARTIC OF $C_{M_{DV}}$

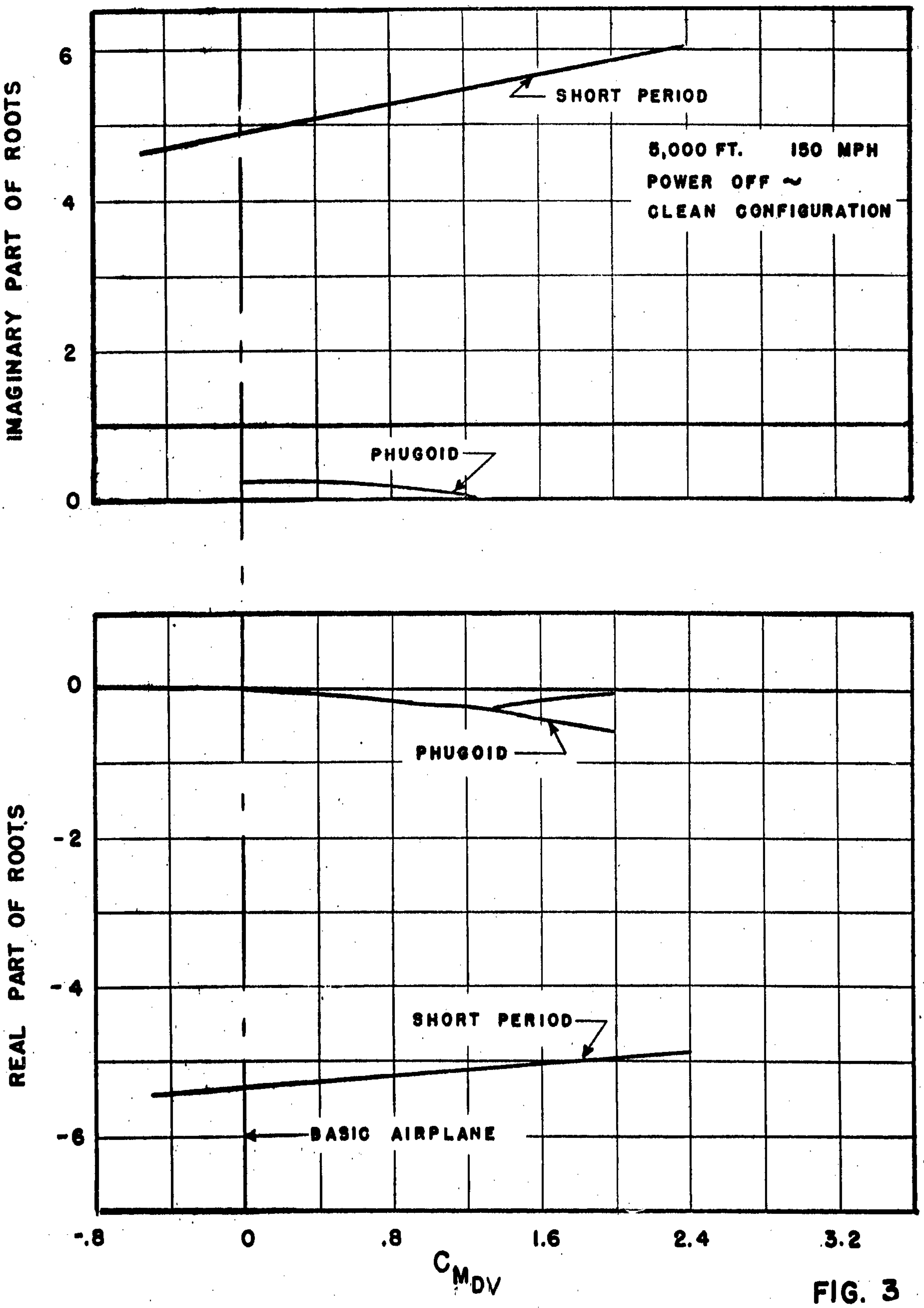


FIG. 3

C-45F

VELOCITY AND NORMAL ACCELERATION RESPONSES TO AN ELEVATOR STEP DEFLECTION

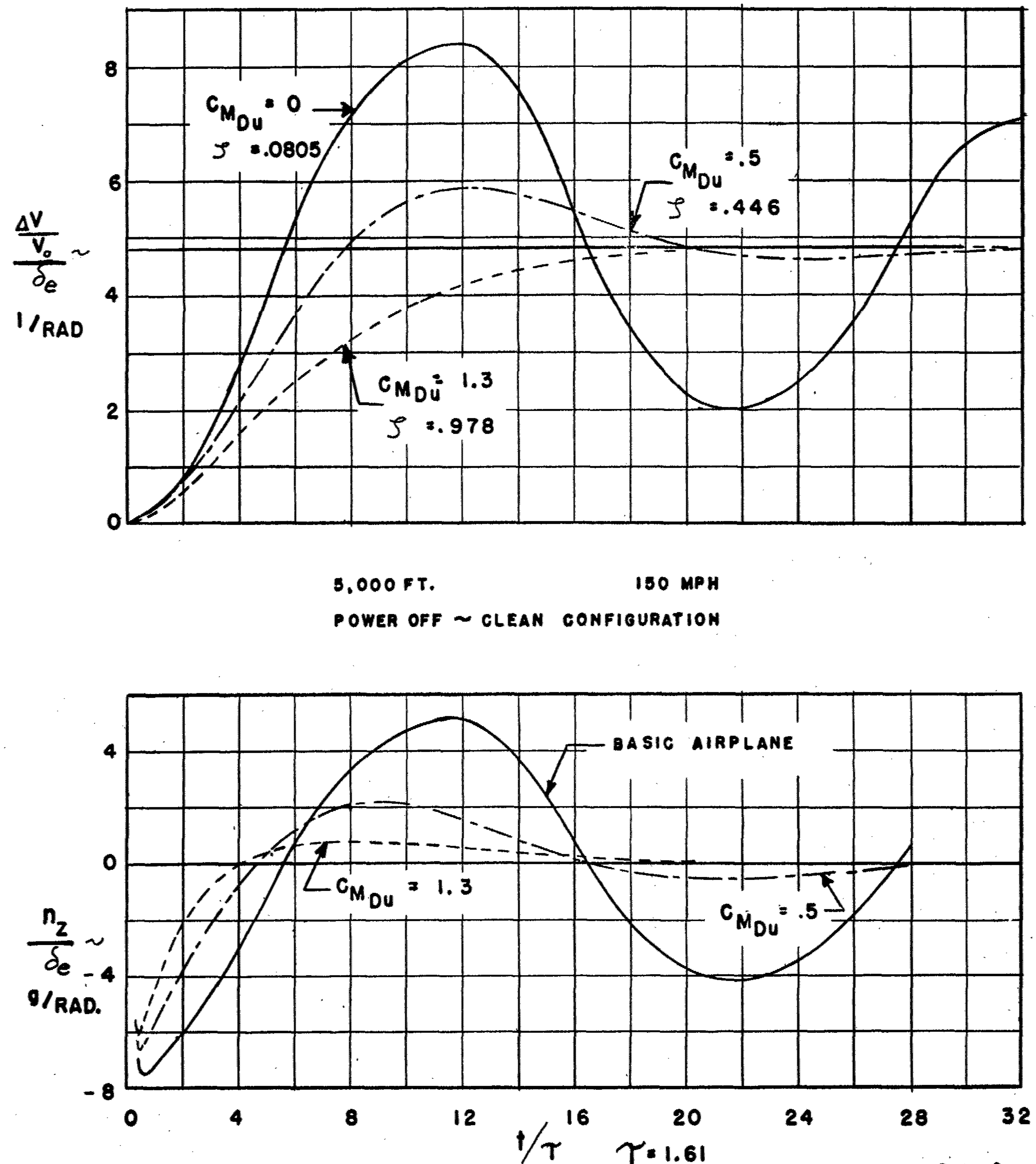


FIG. 4

C-45F

EFFECT OF $-C_{n_r}$ ON LATERAL STABILITY

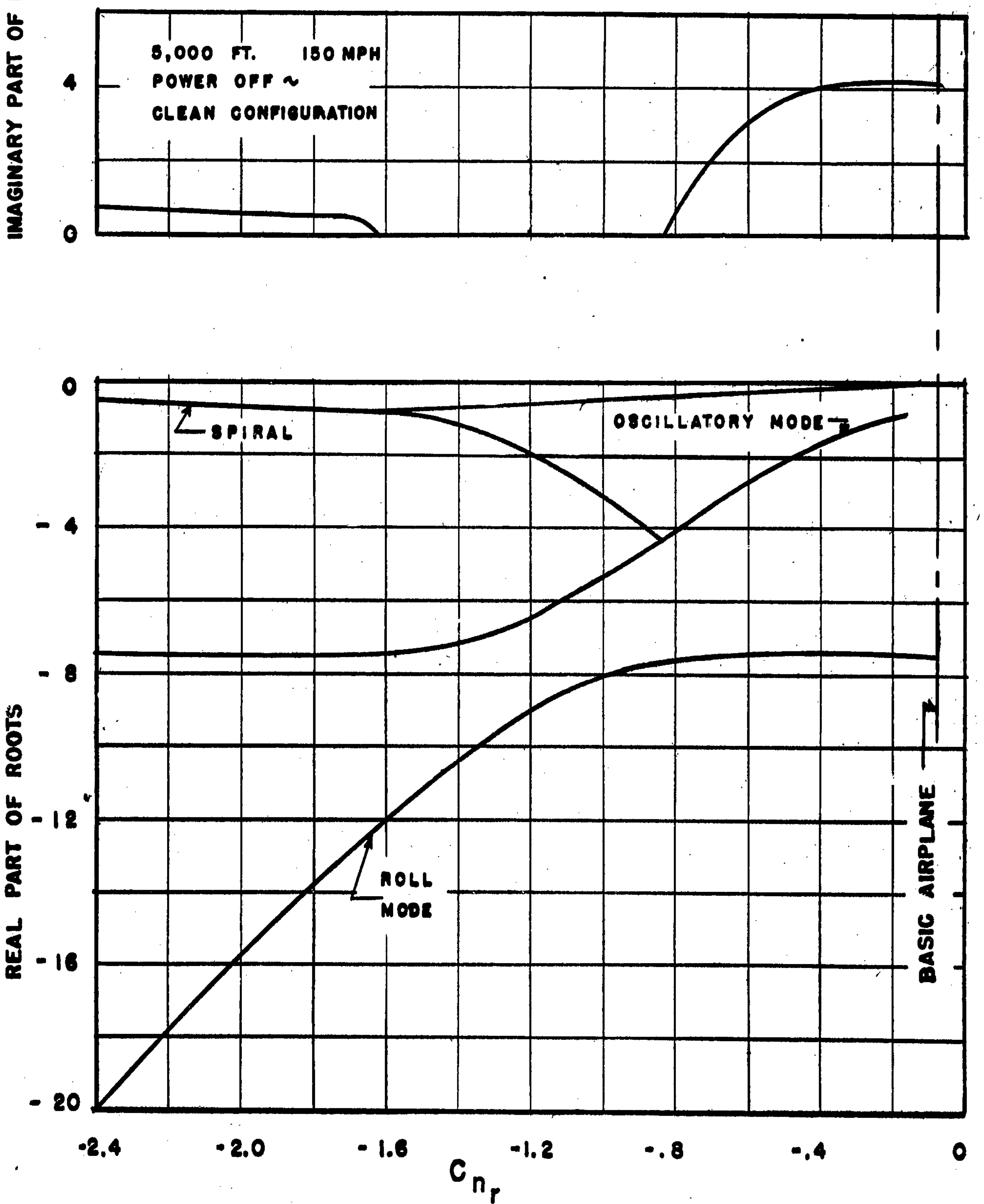


FIG. 5

C-45F

EFFECT OF $C_{n_{Dr}}$ ON LATERAL STABILITY

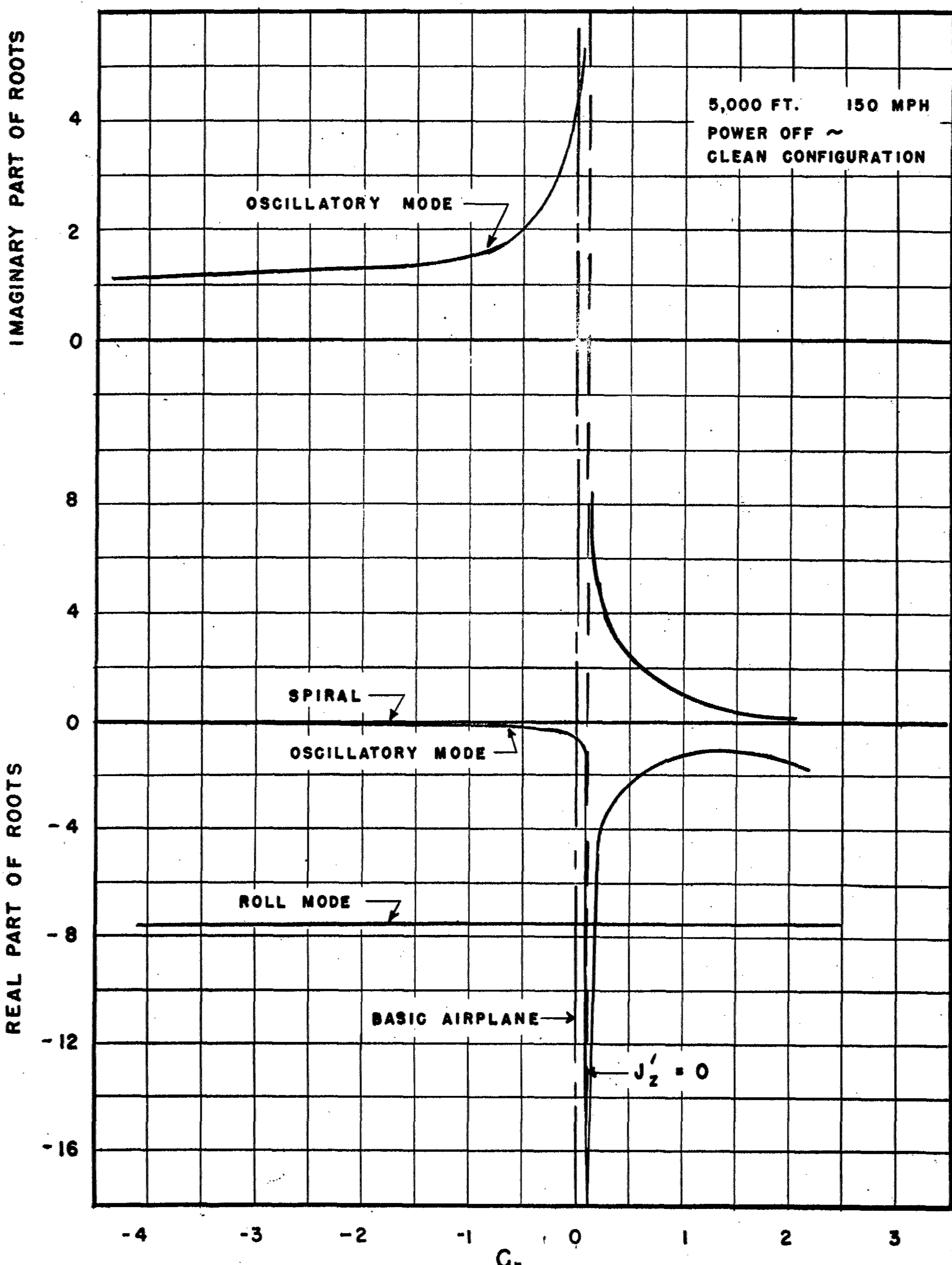


FIG. 6

C-45 F

EFFECT OF $C_{n_{D\beta}}$ ON LATERAL STABILITY

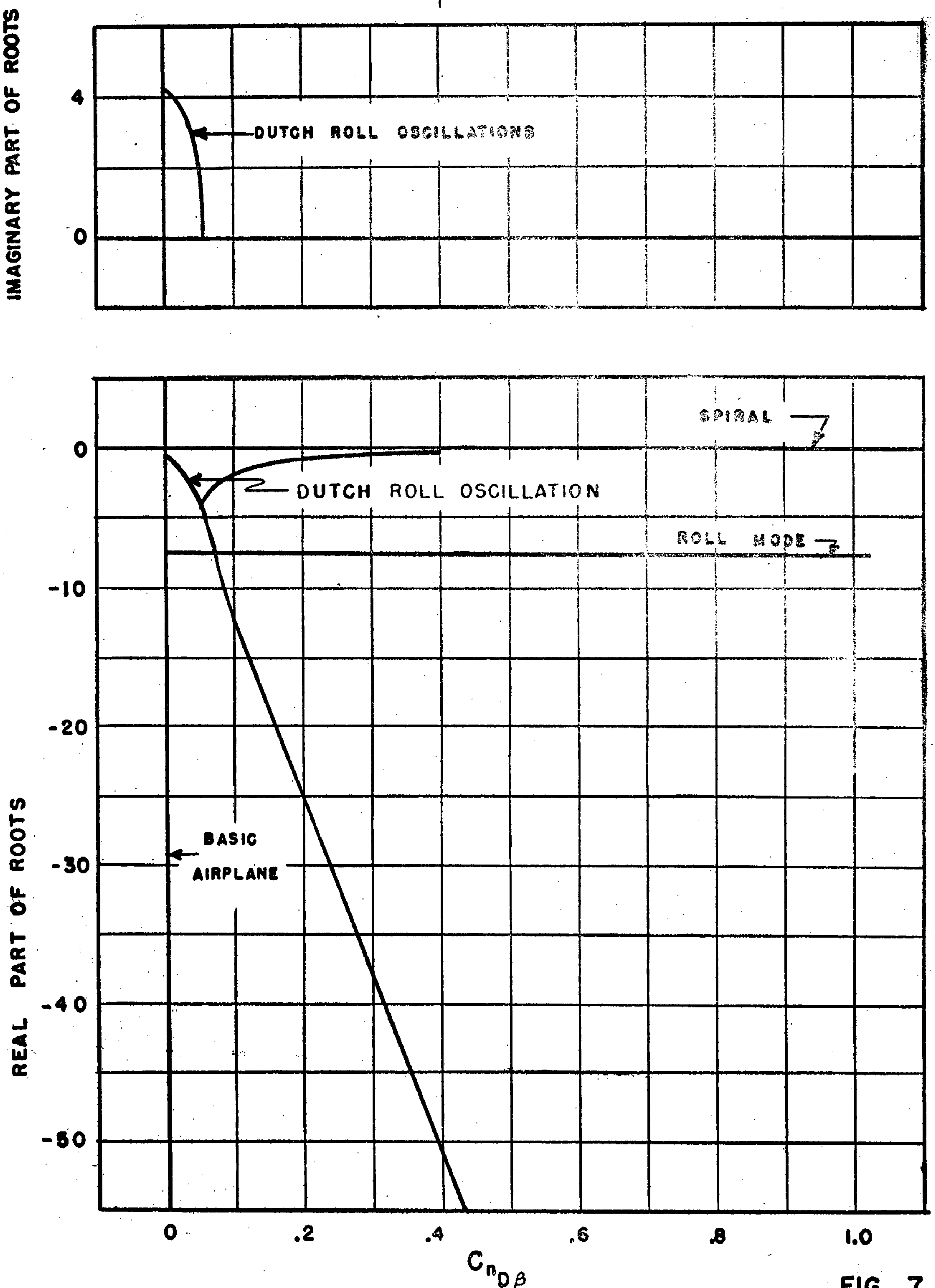


FIG. 7

C-45F

EFFECT OF C_{ℓ_r} ON LATERAL STABILITY

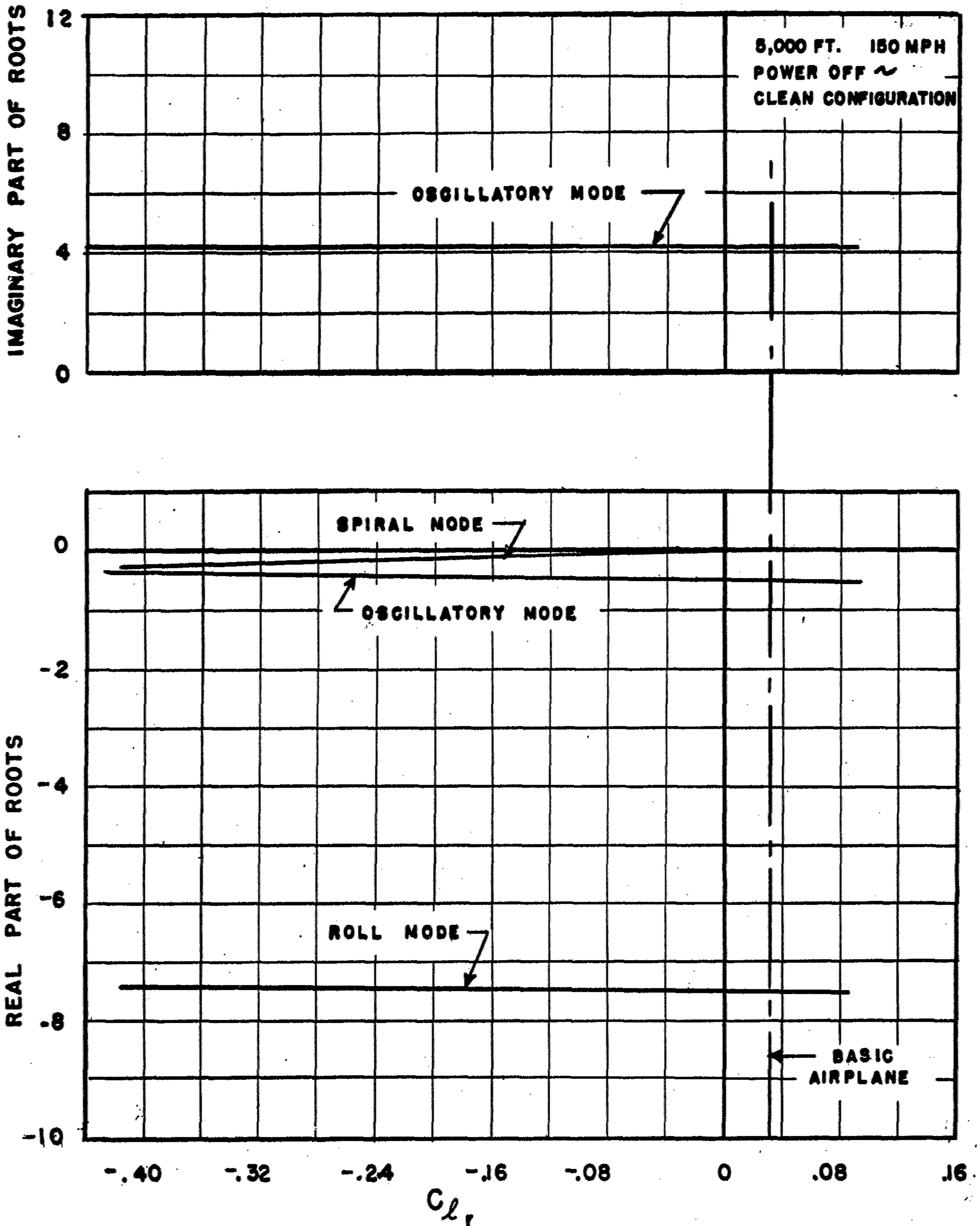


FIG. 8

C - 45F

EFFECT OF $C_{l_{Dp}}$ ON LATERAL STABILITY

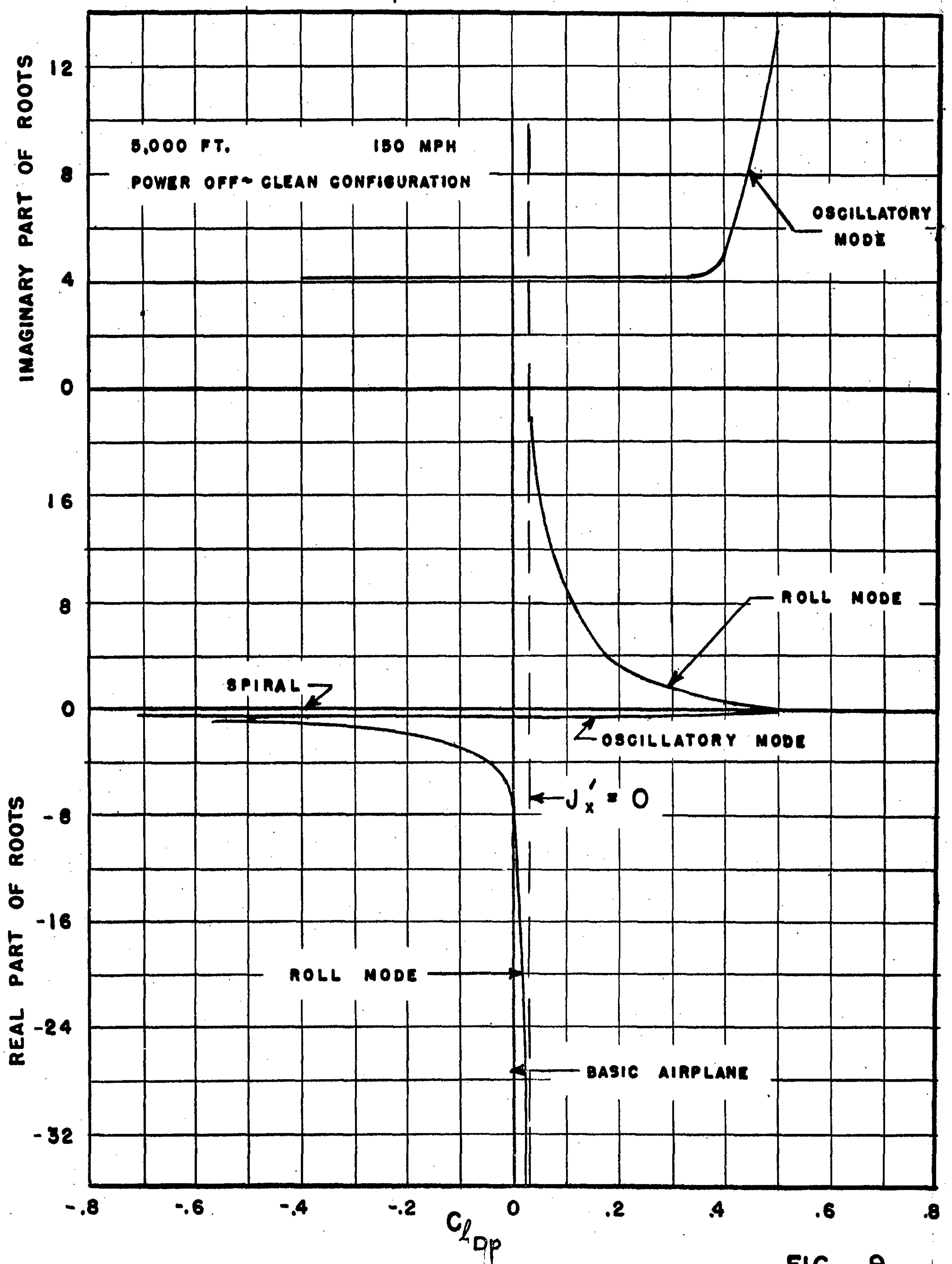


FIG. 9

C - 45 F

ROLL RATE RESPONSES TO STEP AND EXPONENTIAL RUDDER AND AILERON DEFLECTION $\frac{pb/2V}{\delta_a}$, $\frac{pb/2V}{\delta_r}$

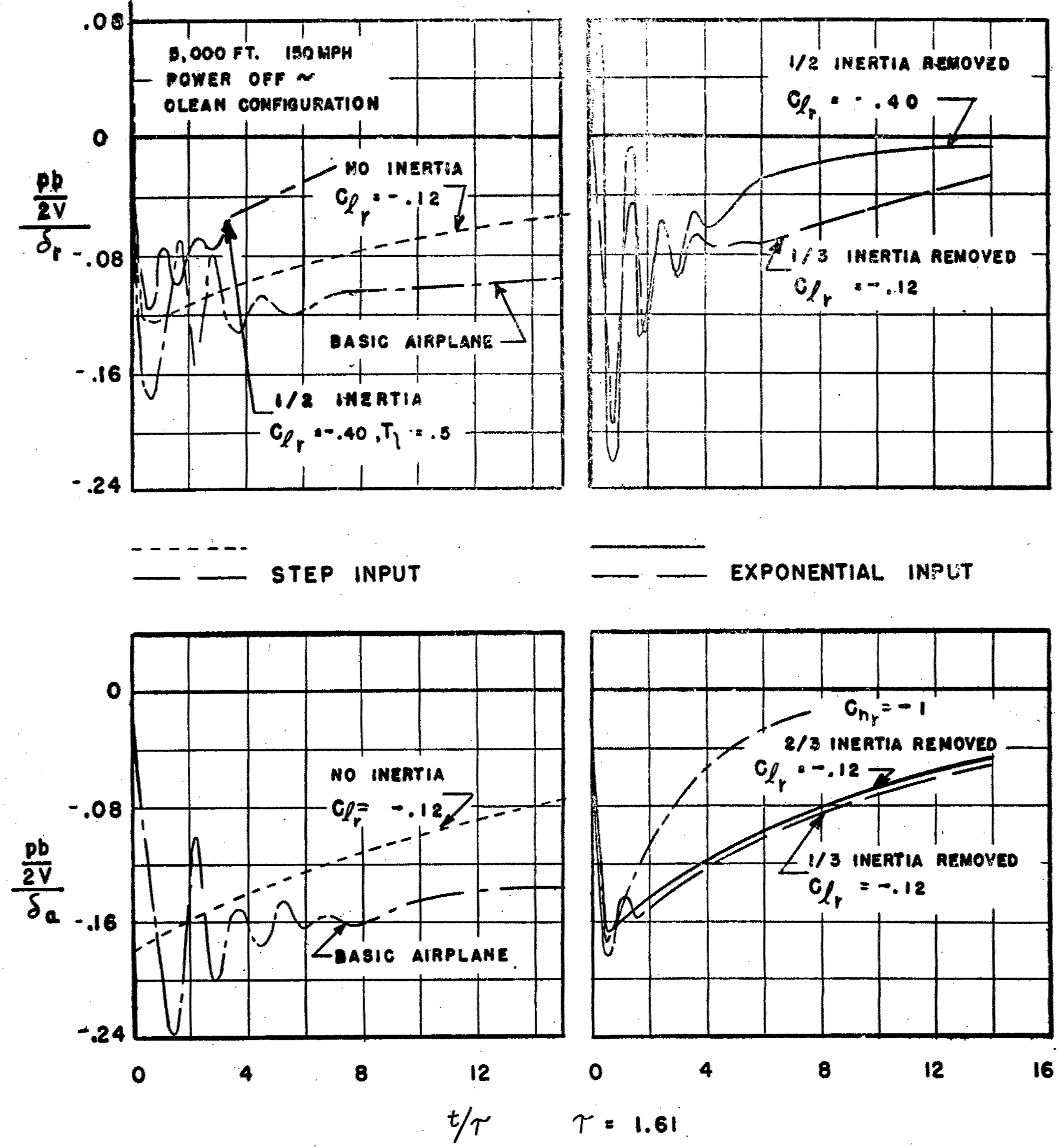


FIG. 10

C-45F

SIDESLIP RESPONSES TO STEP AND EXPONENTIAL RUDDER AND AILERON DEFLECTION β/δ_r, β/δ_a

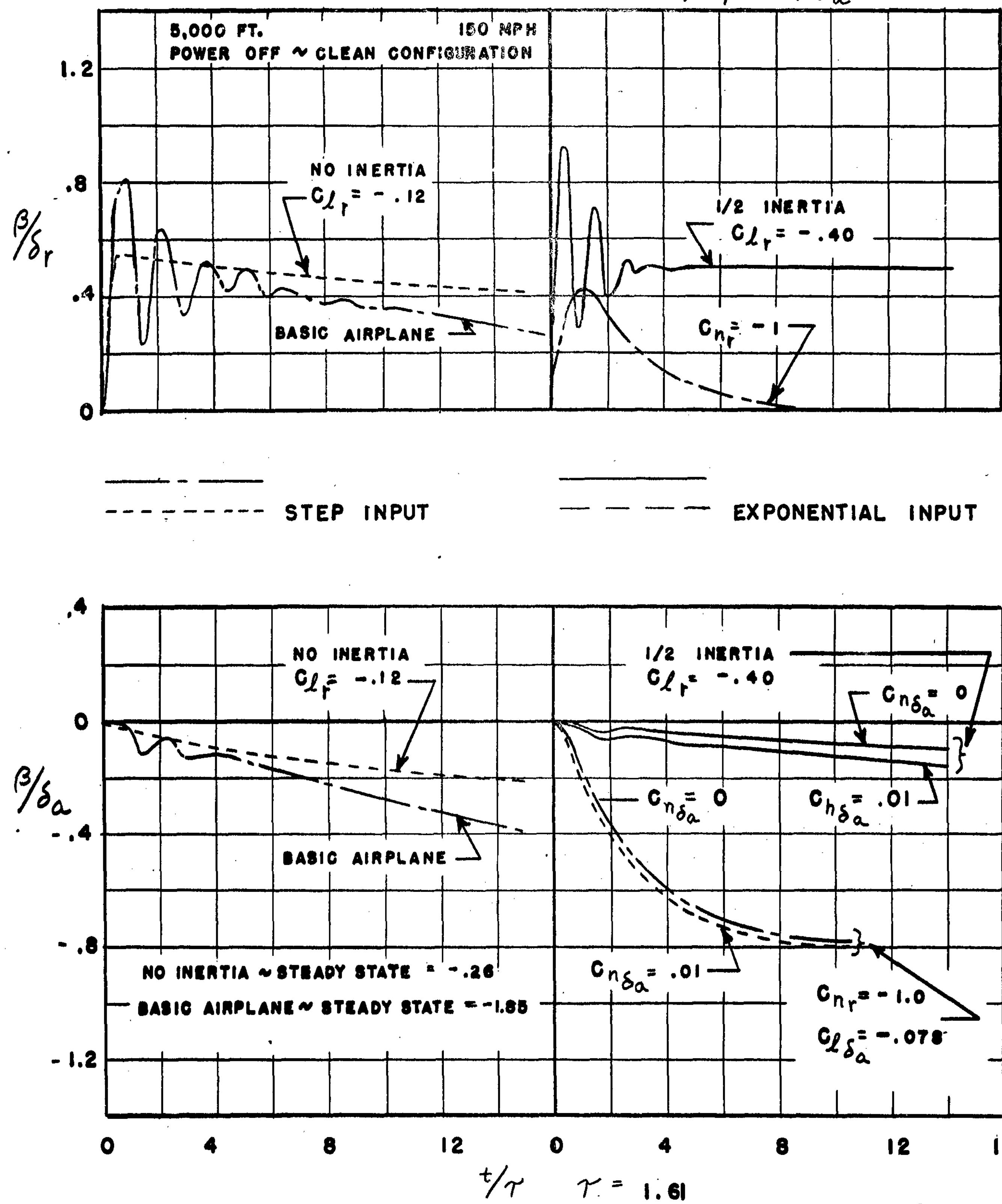
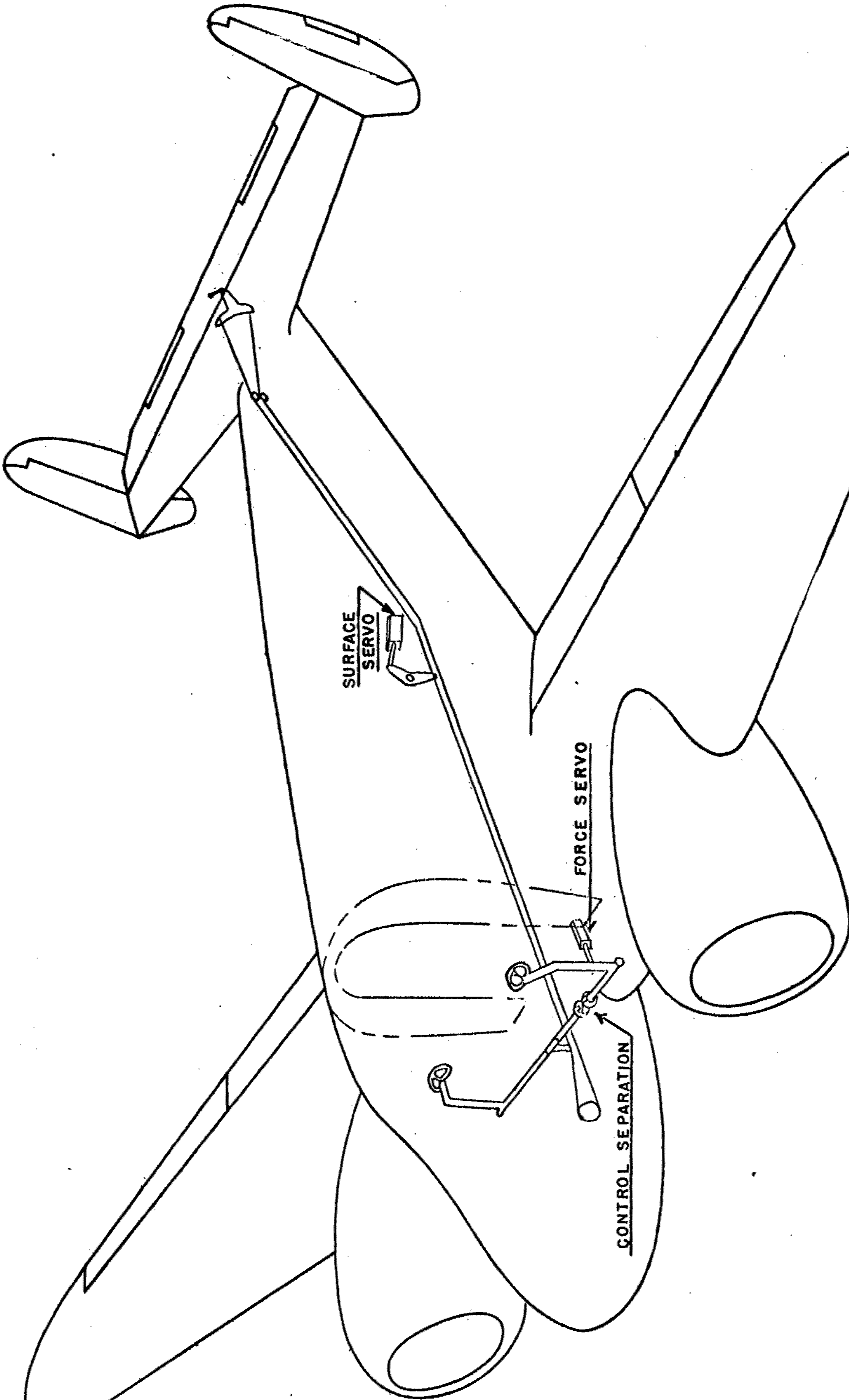
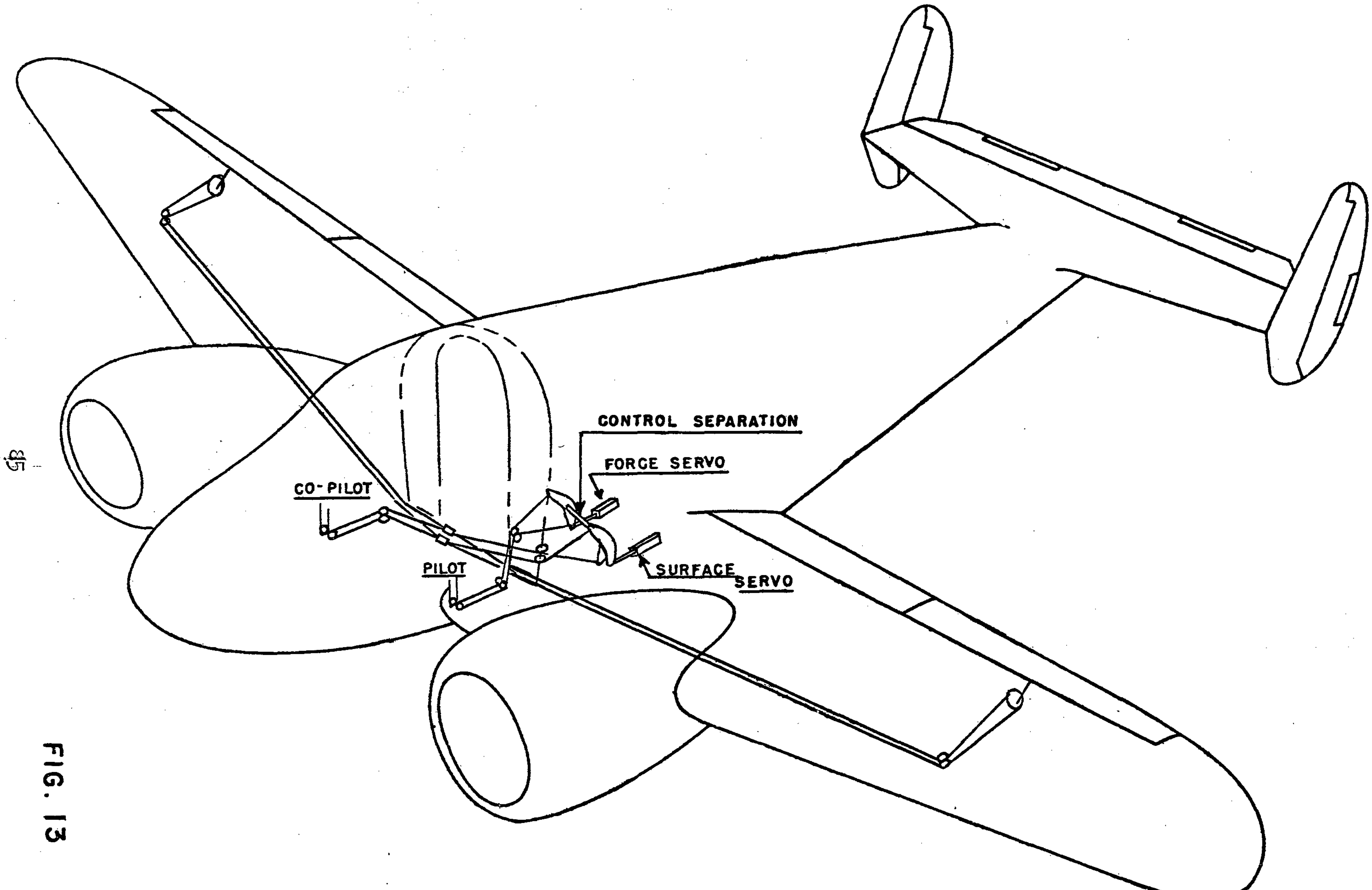


FIG. 11



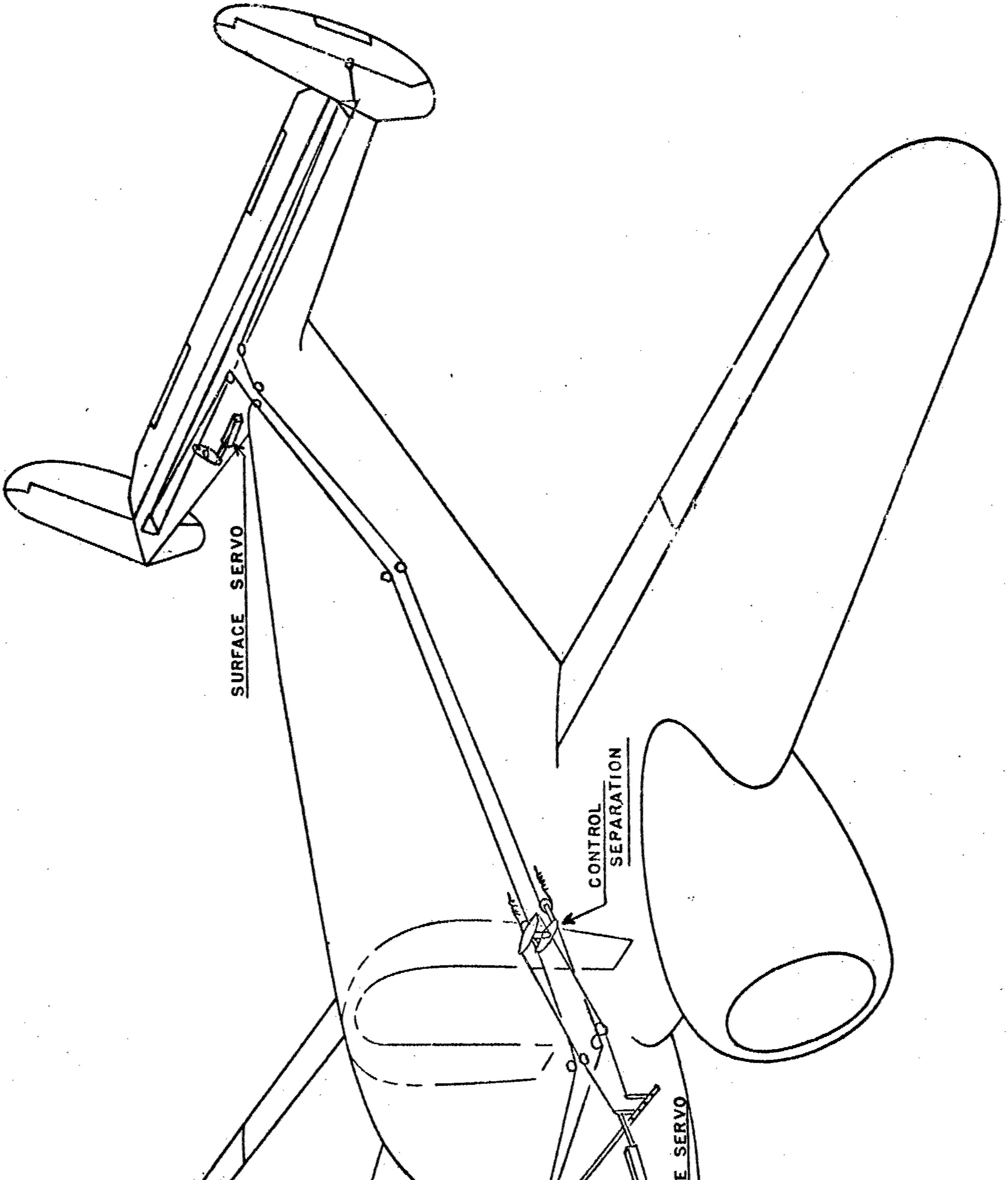
MODIFIED ELEVATOR CONTROL SYSTEM

FIG. 12



MODIFIED AILERON CONTROL SYSTEM

FIG. 13



MODIFIED RUDDER CONTROL SYSTEM

FIG. 14

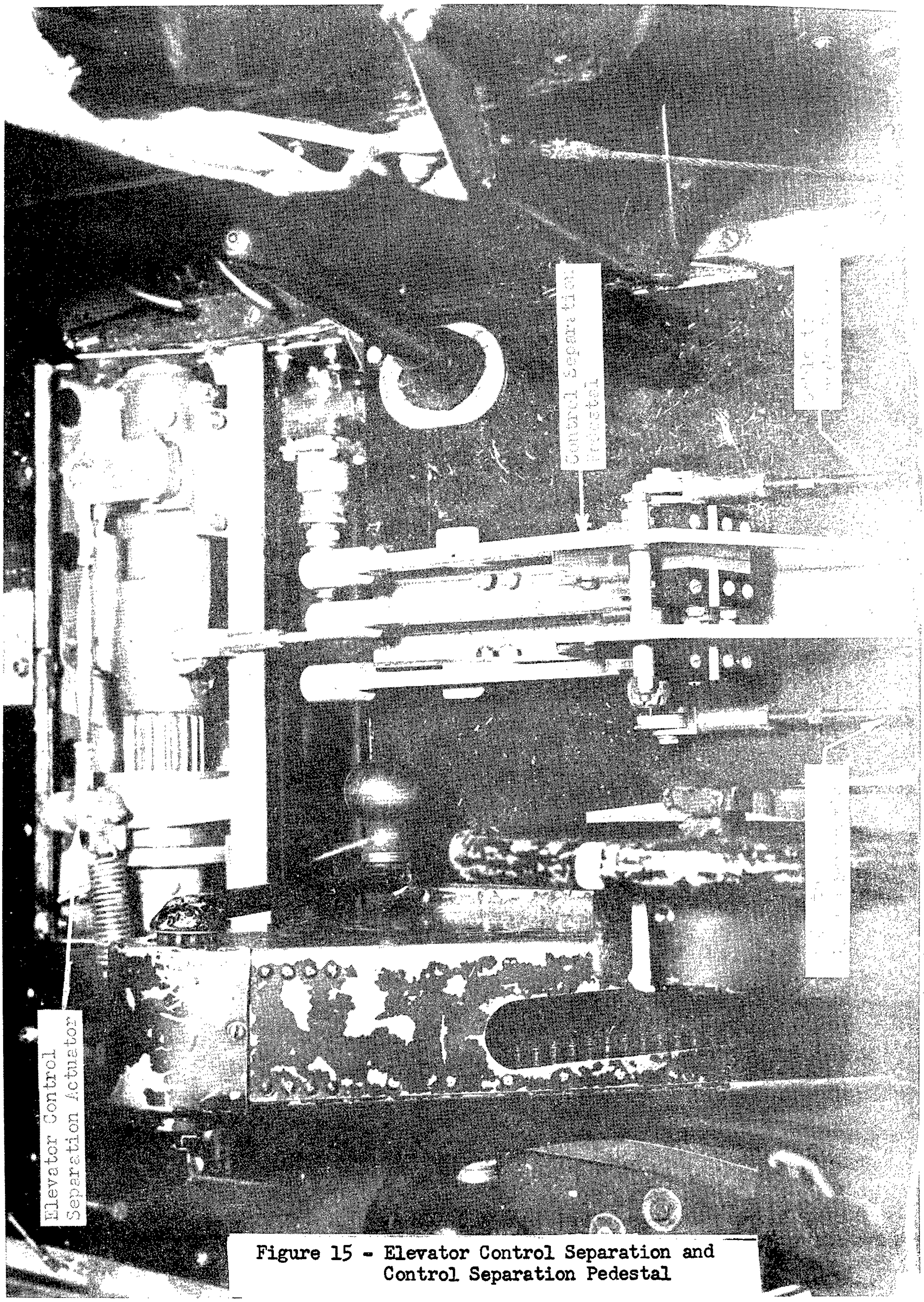


Figure 15 - Elevator Control Separation and Control Separation Pedestal

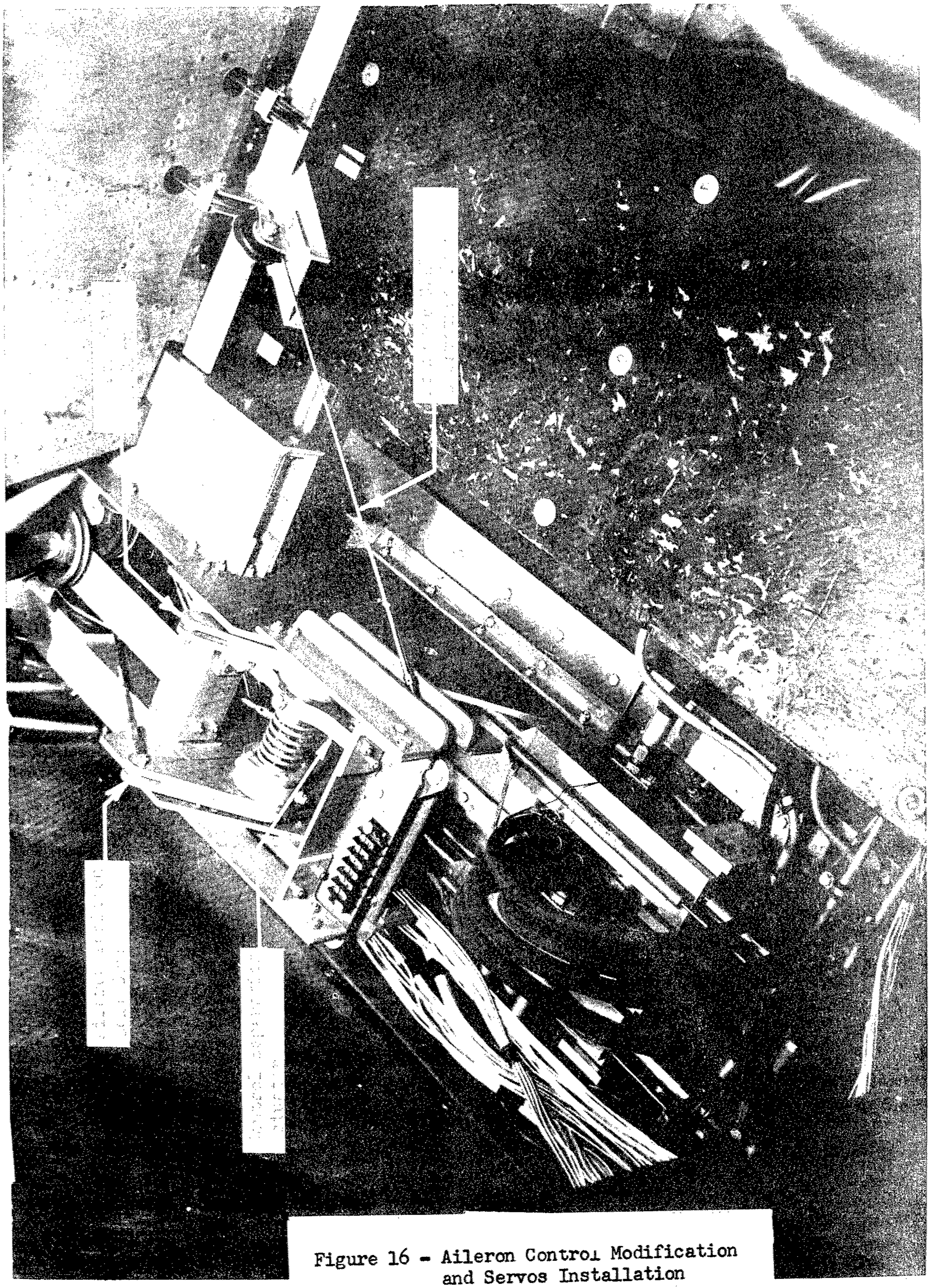


Figure 16 - Aileron Control Modification and Servos Installation

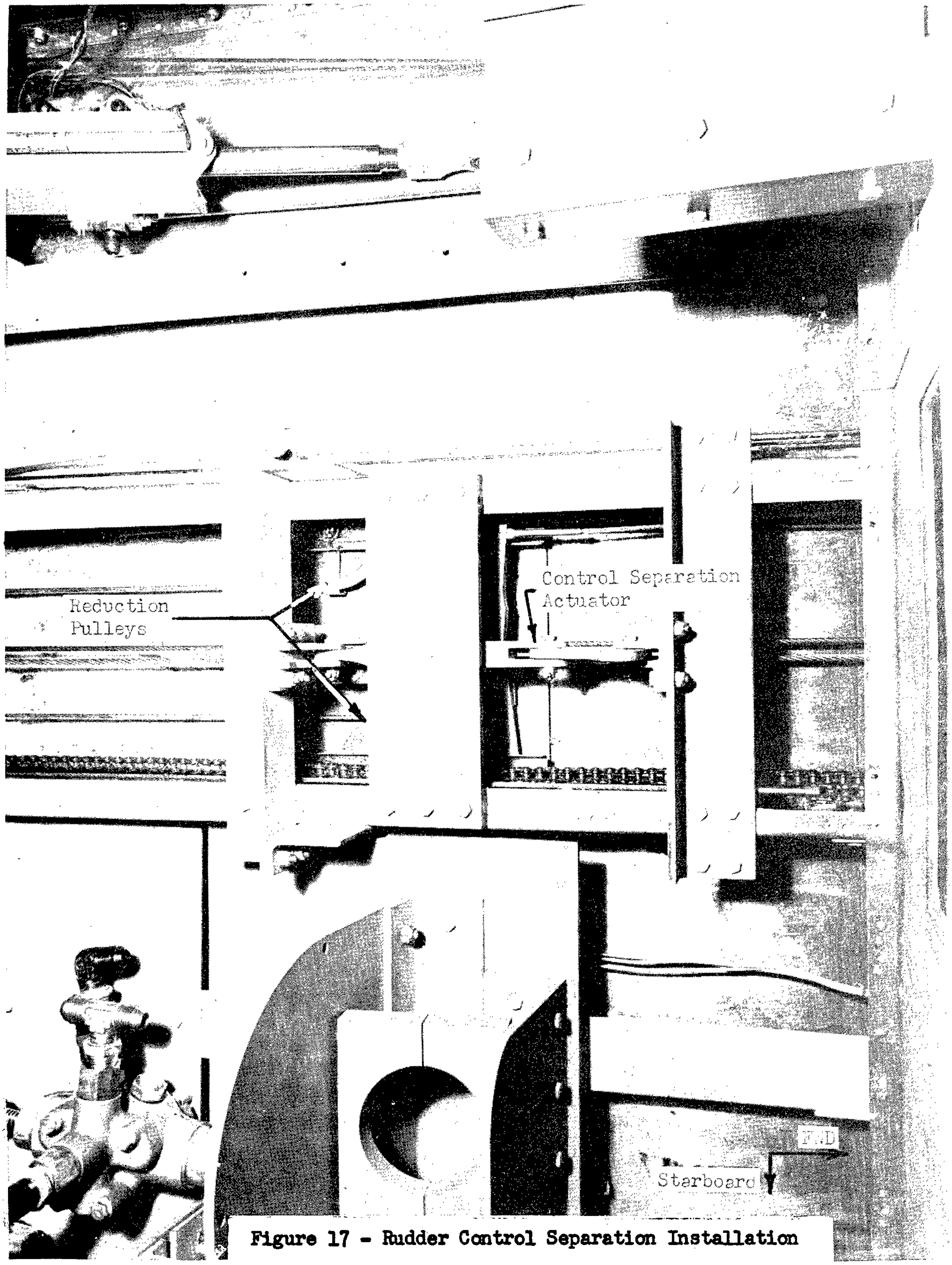


Figure 17 - Rudder Control Separation Installation

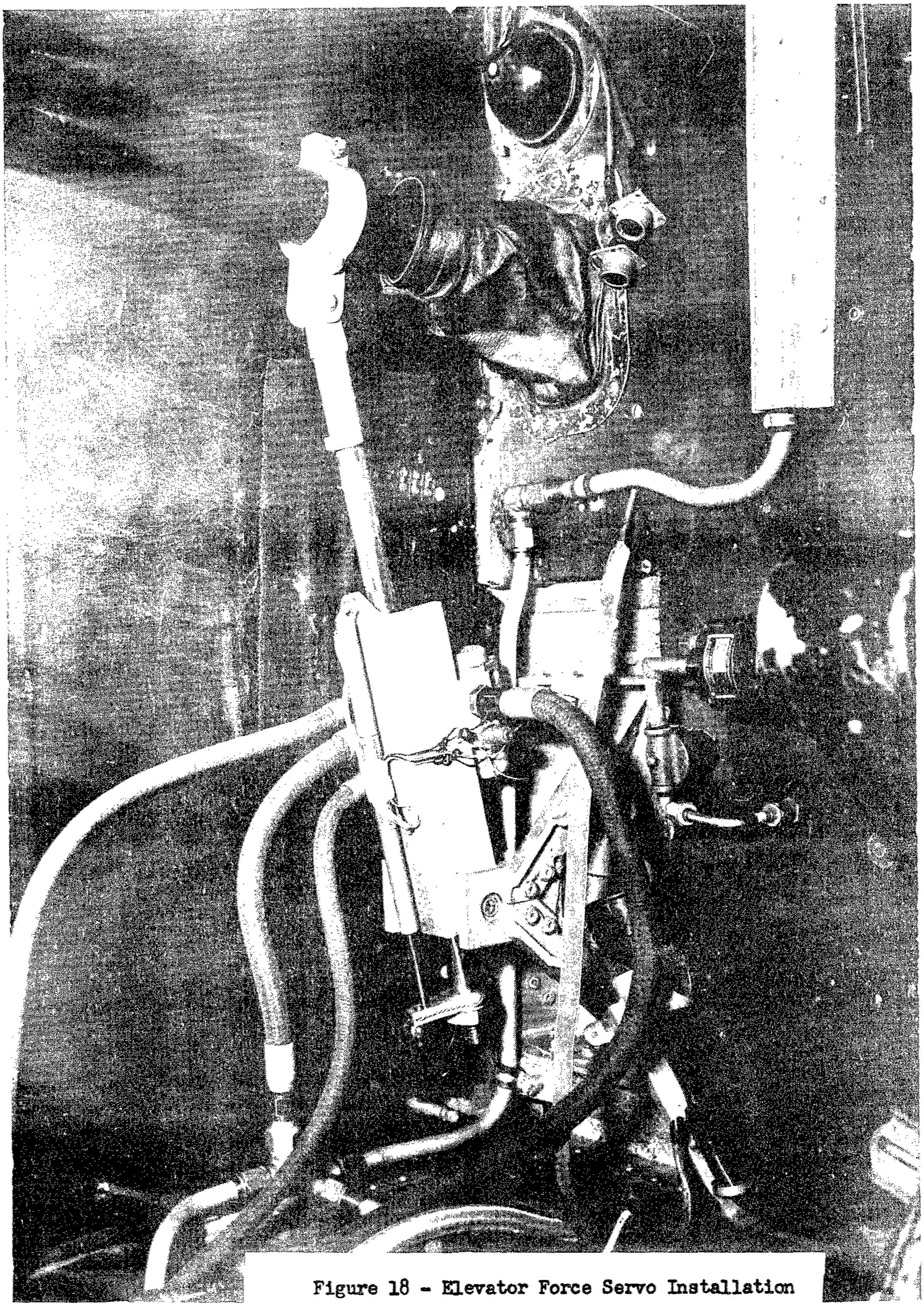


Figure 18 - Elevator Force Servo Installation

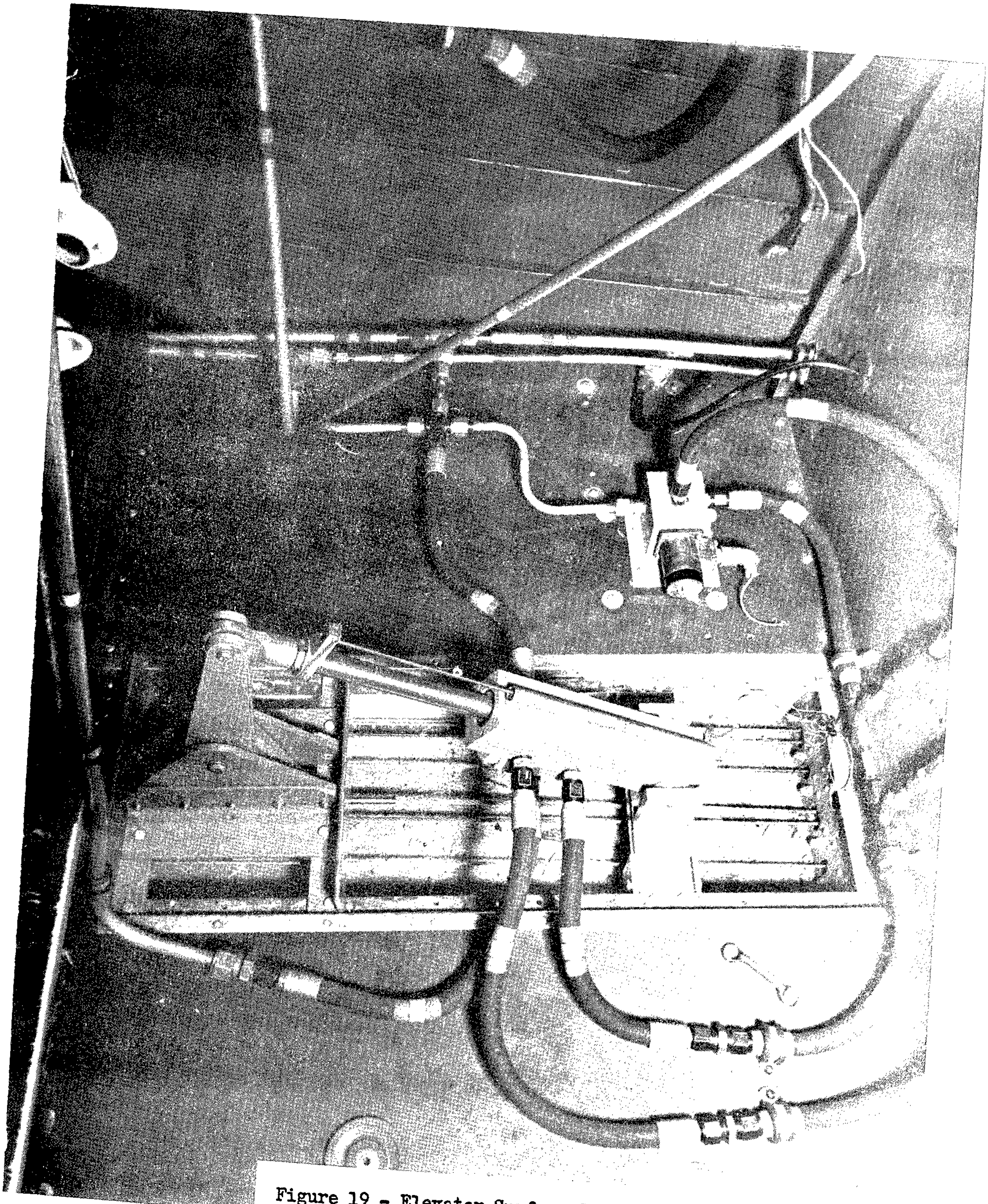


Figure 19 - Elevator Surface Servo Installation

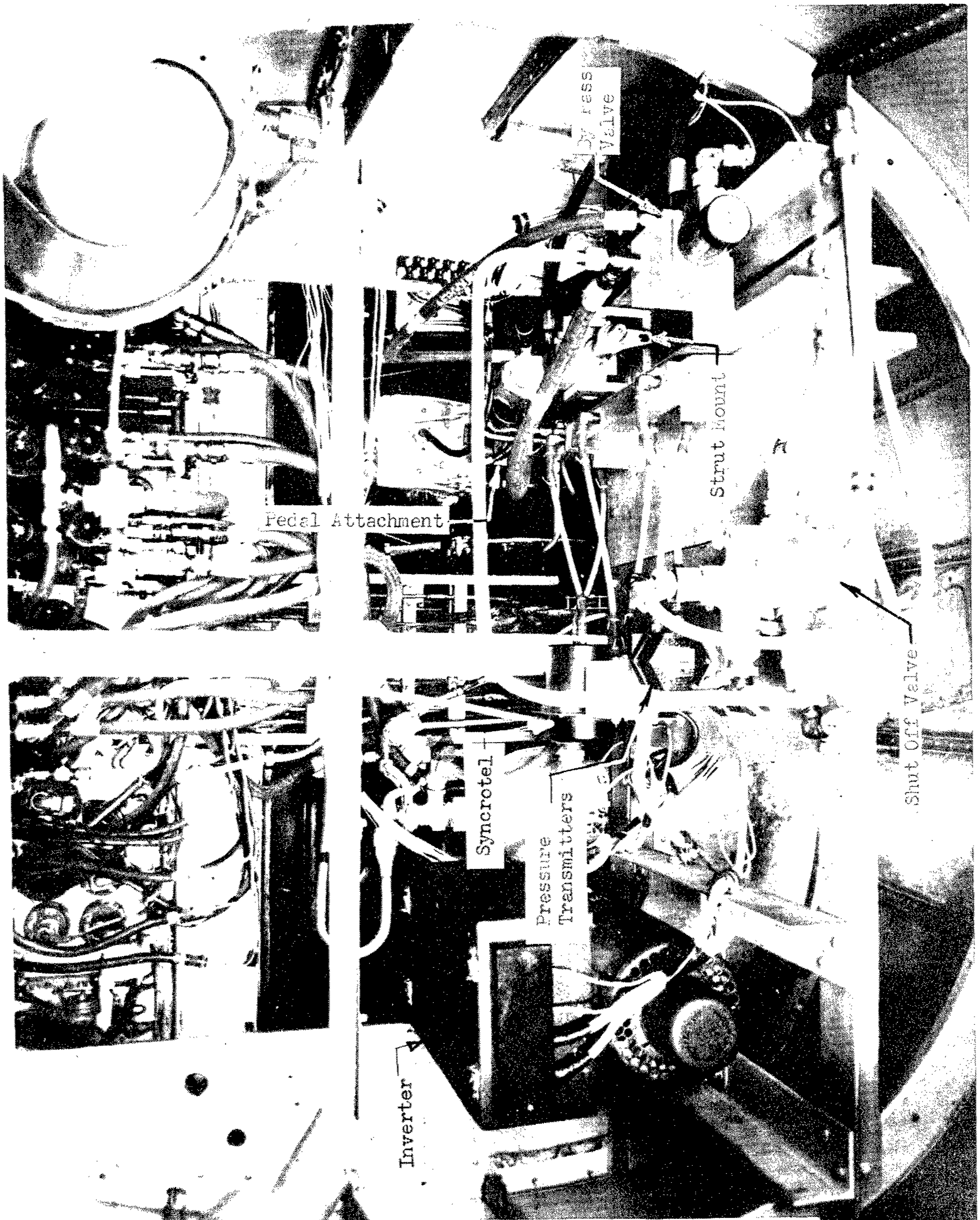


Figure 20 - Rudder Force Servo Installation In Nose Compartment

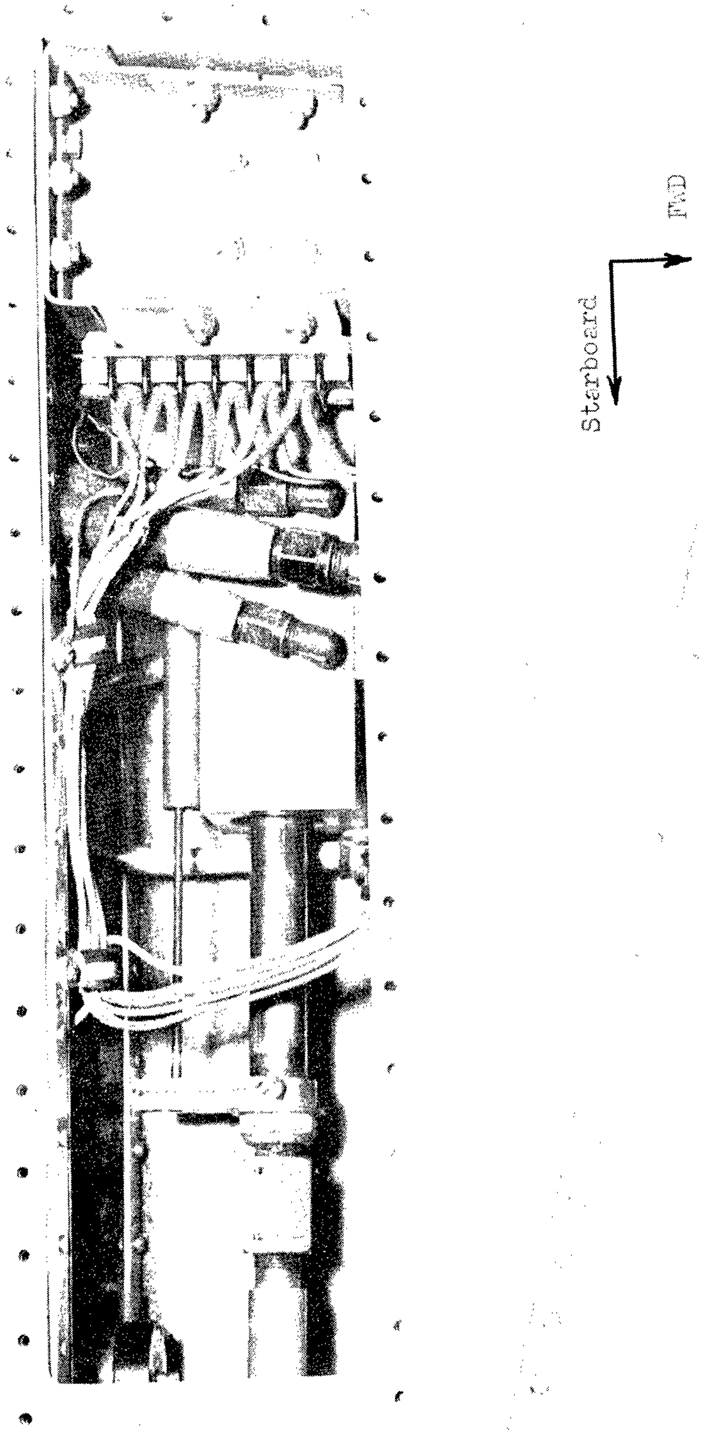


Figure 21 - Rudder Surface Servo Installation
In Horizontal Stabilizer

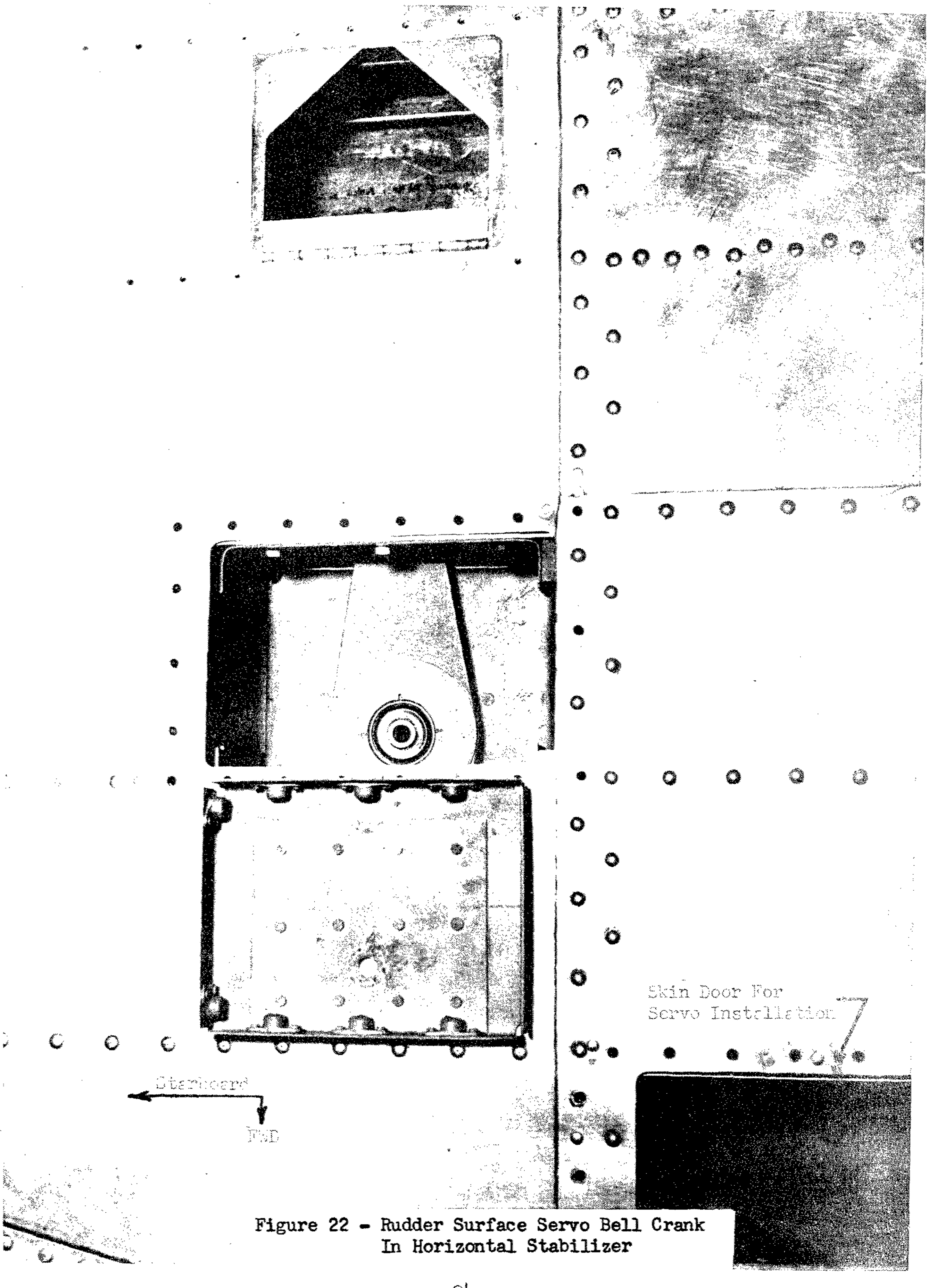


Figure 22 - Rudder Surface Servo Bell Crank
In Horizontal Stabilizer

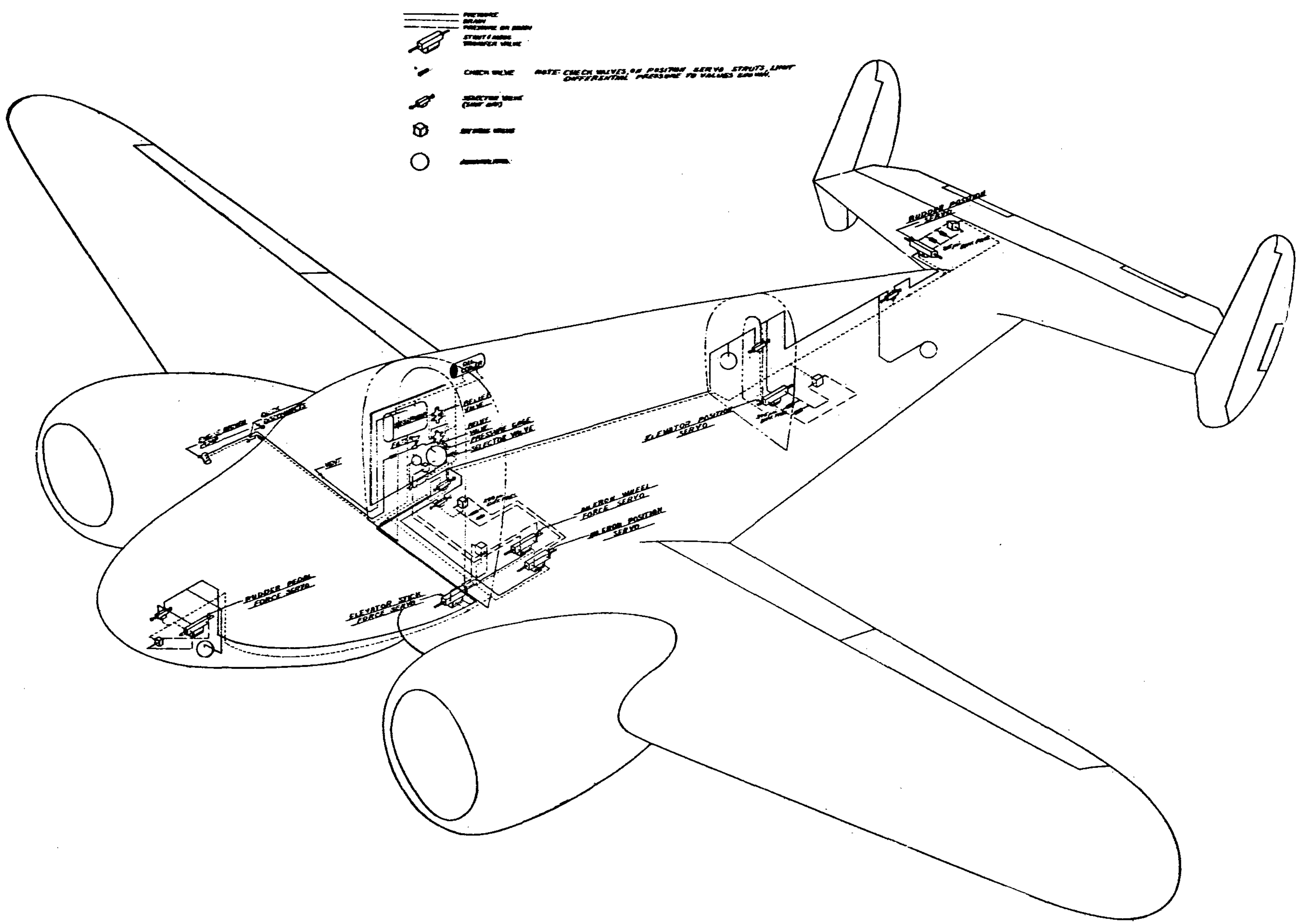
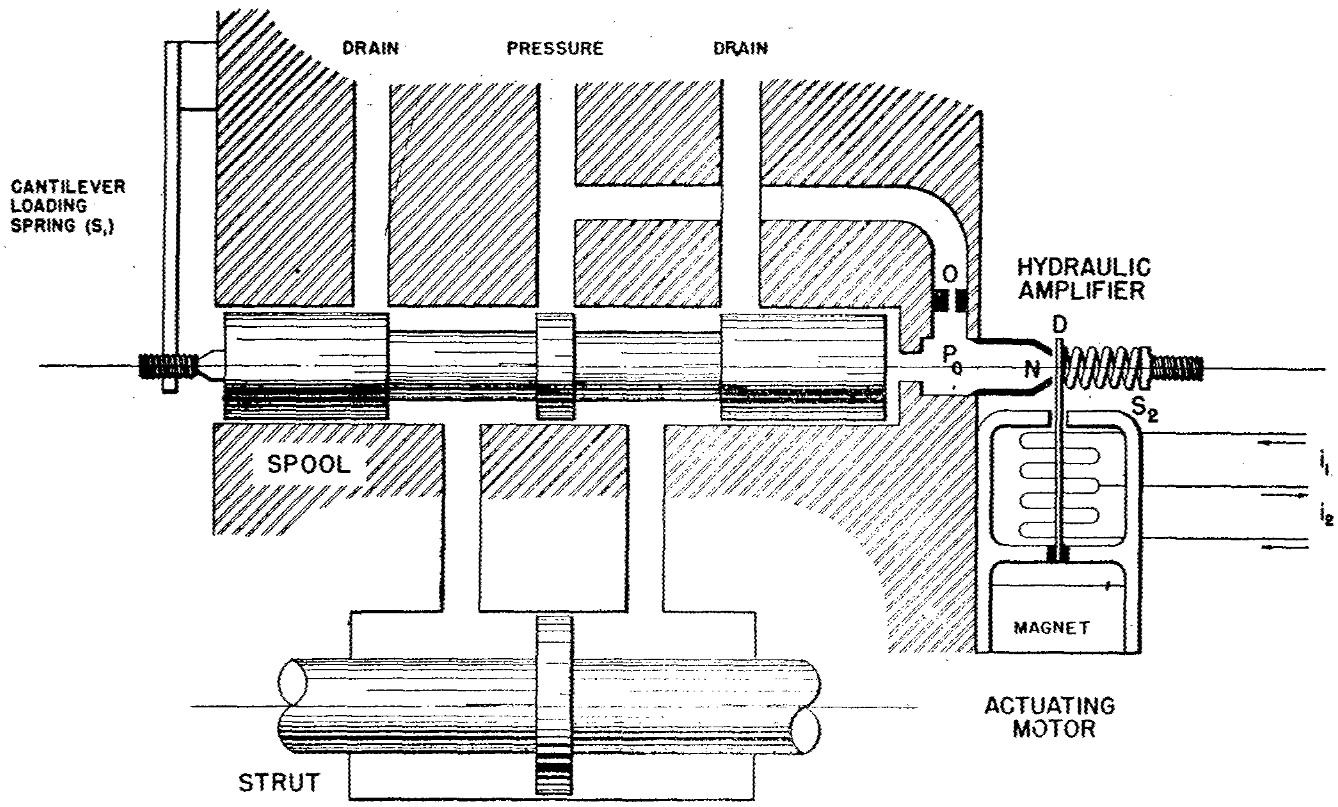


Fig. 23 - HYDRAULIC SYSTEM SCHEMATIC

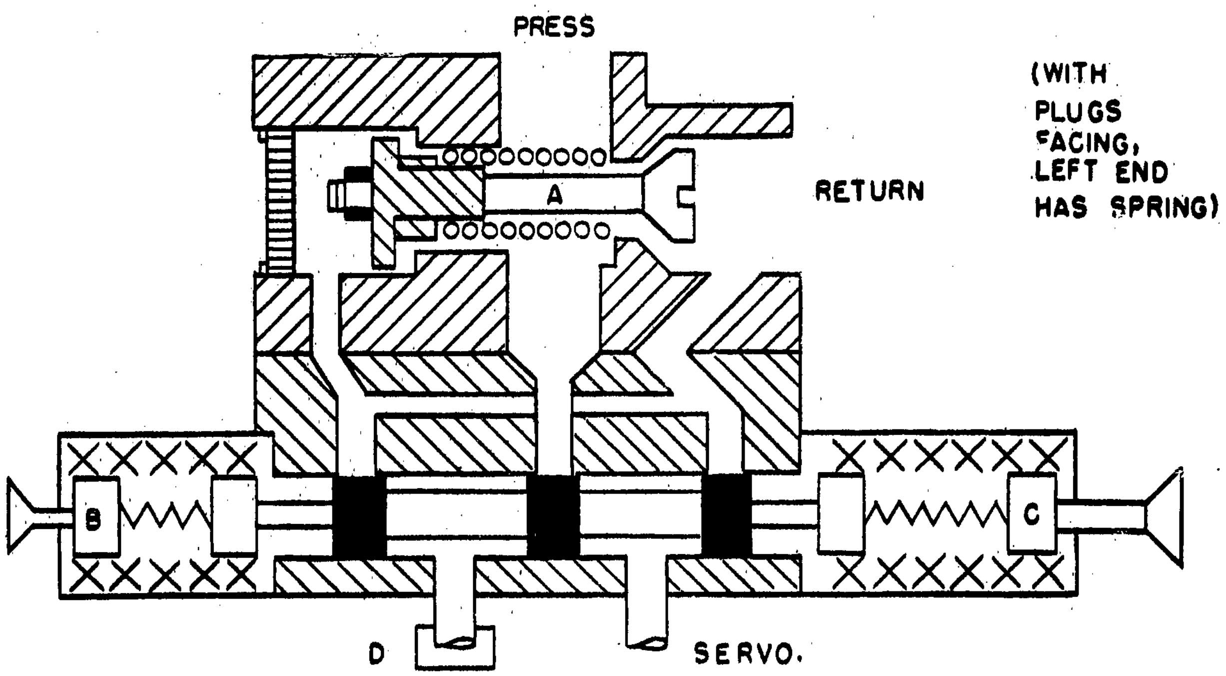


SCHEMATIC

CORNELL DIRECT CURRENT NOZZLE DRIVE
TRANSFER VALVE

FIG. 24

SHUT-OFF VALVE MODIFICATION



1. "A" assembly tightened to prevent flow from P to R when solenoids are degenerated and P is at system pressure.
2. Spring at "C" discarded and spring at "B" replaced by same length spring having lower rate.
3. Port at "D" sealed.
4. Use solenoid at "B" for actuation.
5. Spool at rest off center toward "C".

   Spool at "ON" off center toward "B".

FIG. 25

HYDRAULIC SUPPLY SYSTEM CONTROLS

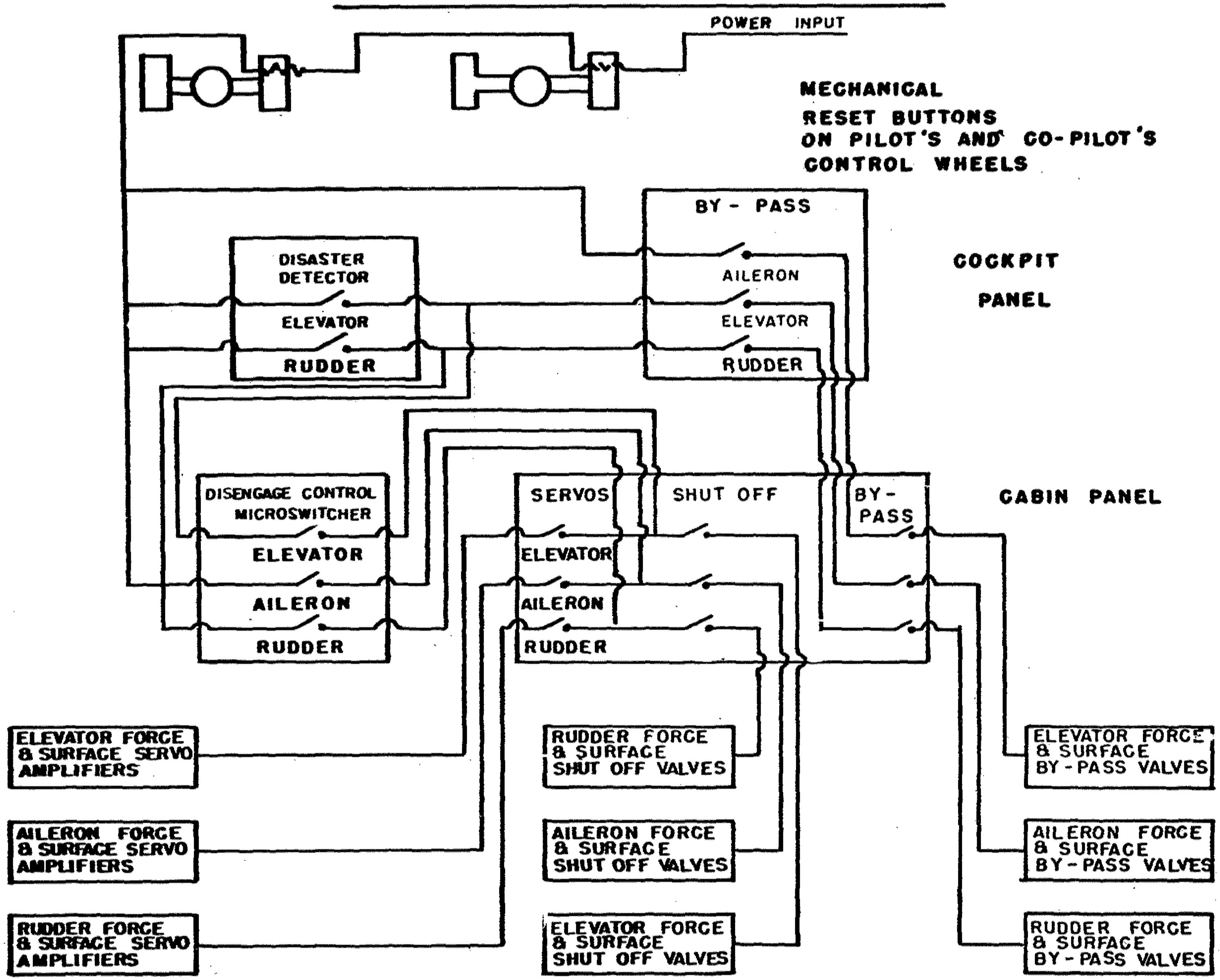


FIG. 26

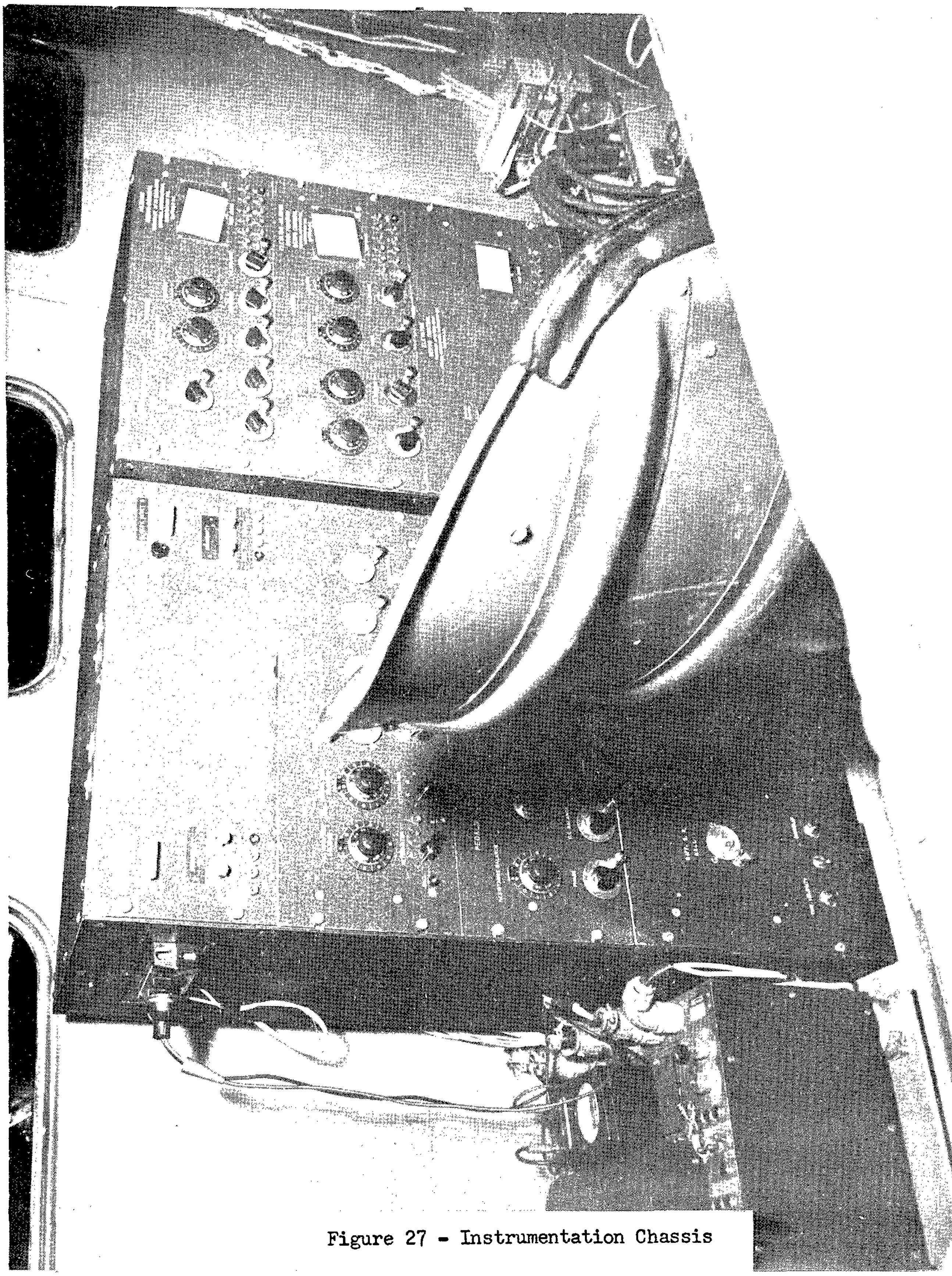


Figure 27 - Instrumentation Chassis

Figure 28 - C-45 Cabin Looking Aft

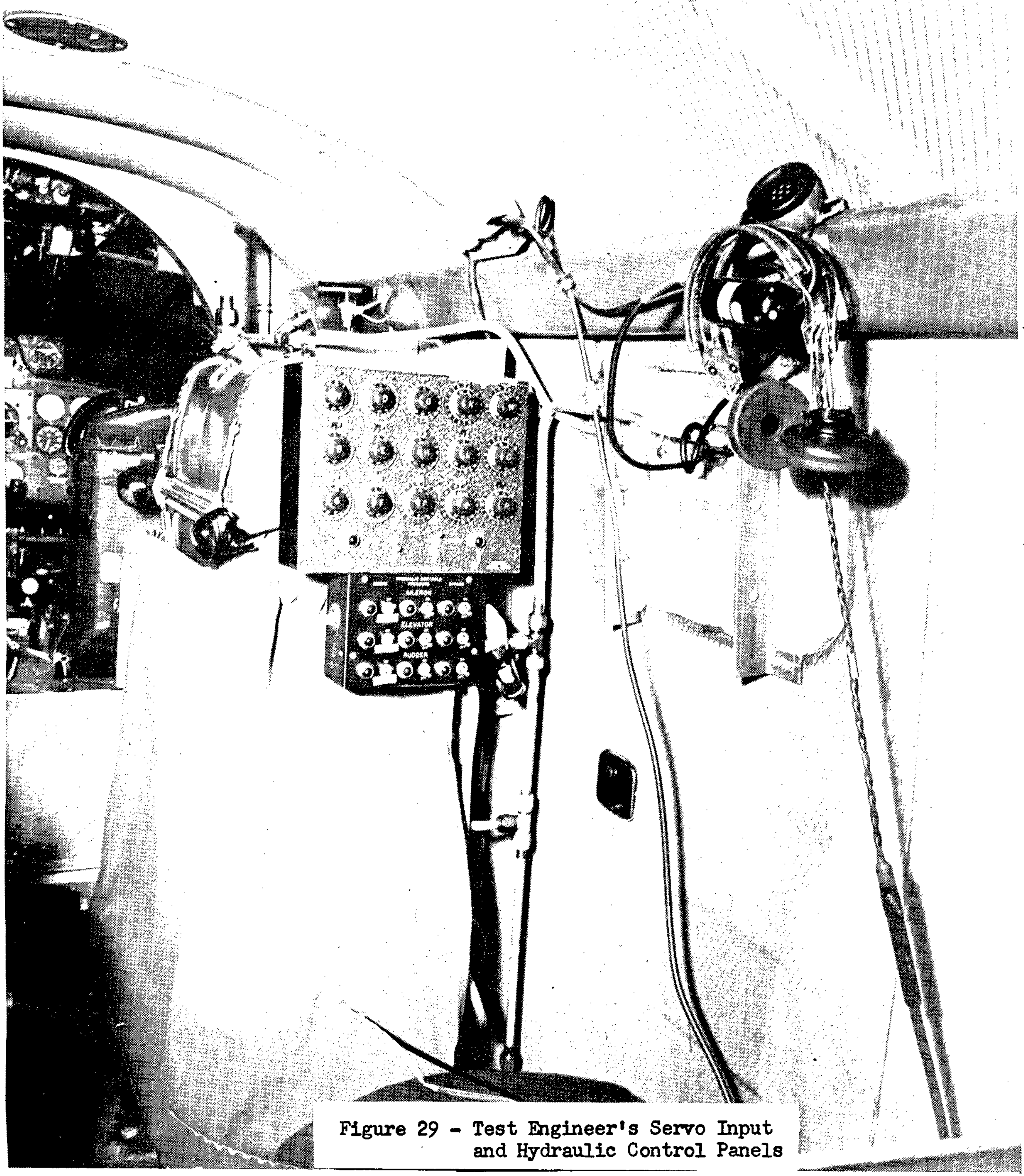


Figure 29 - Test Engineer's Servo Input and Hydraulic Control Panels

DISASTER DETECTOR

1. Power reset switch
2. Rudder disaster reset
3. Rudder disaster sensitivity adj.
4. Elevator disaster reset
5. Elevator disaster sensitivity adj.

RECORDING CONTROL

1. Airspeed balance
2. Airspeed attenuator
3. Vertical accel. balance
4. Vertical accel. attenuator
5. Rudder position null
6. Rudder position attenuator
7. Aileron position null
8. Aileron position attenuator
9. Elevator position null
10. Elevator position attenuator
11. Sideslip null (recording only)
12. Sideslip attenuator

RUDDER SERVO CONTROL

1. Pedal force balance (resistance)
2. Pedal force balance (phase)
3. Pedal position balance
4. D.C. balance (pedal transfer valve)
5. Yawing accelerometer (balance)
6. Yawing acceleration recording attenuator
7. Surface position balance
8. Cross over feed (surface positioning from force servo) labeled "Feedback"
9. D.C. balance (surface trans. valve)

POWER AND RUDDER SERVO CONTROL

1. Differentiated sideslip (null)
2. Diff. sideslip recording atten.
3. Sideslip (null)
4. Yaw velocity - sideslip to rudder selector switch

AILERON SERVO CONTROL

1. Rolling accelerometer balance (resistance)
2. Rolling accelerometer balance (phase)
3. Roll Accel. recording attenuator
4. Surface position balance
5. Cross over feed (surface positioning from force servo) labeled "Feedback"
6. D.C. balance (surface trans. valve)
7. Yaw angular velocity recording attenuator
8. Roll angular velocity recording attenuator

AILERON AND ELEVATOR STICK SERVO CONTROL

1. Elevator force balance (resistance)
2. Elevator force balance (phase)
3. Elevator stick position balance
4. D.C. balance (Elev. stick trans. valve)
5. Aileron force balance (resistance)
6. Aileron force balance (phase)
7. Aileron wheel position balance
8. D.C. balance (aileron stick trans. valve)

ELEVATOR SERVO CONTROL

1. Surface position balance
2. Cross over feed (surface positioning from force servo) labeled "Feedback"
3. Sensitivity differentiated airspeed
4. Damping differentiated airspeed
5. Differentiated airspeed recording attenuator
6. D.C. balance (surface trans. valve)

DIVIDER UNIT

1. Q divider servo sensitivity
2. Yaw velocity to aileron low pass filter selector switch

C - 45F

CONTROL CHASSIS SCHEMATIC

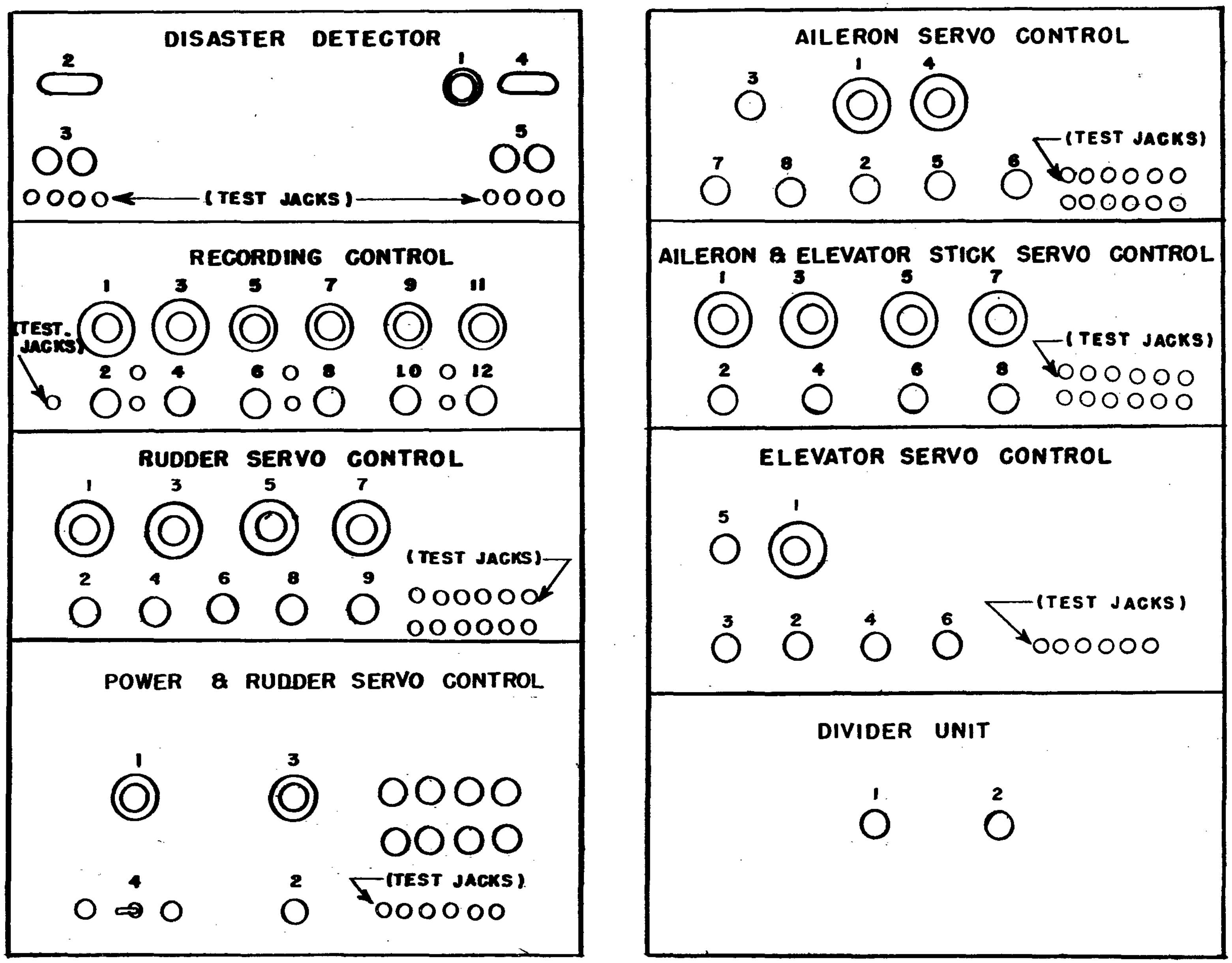
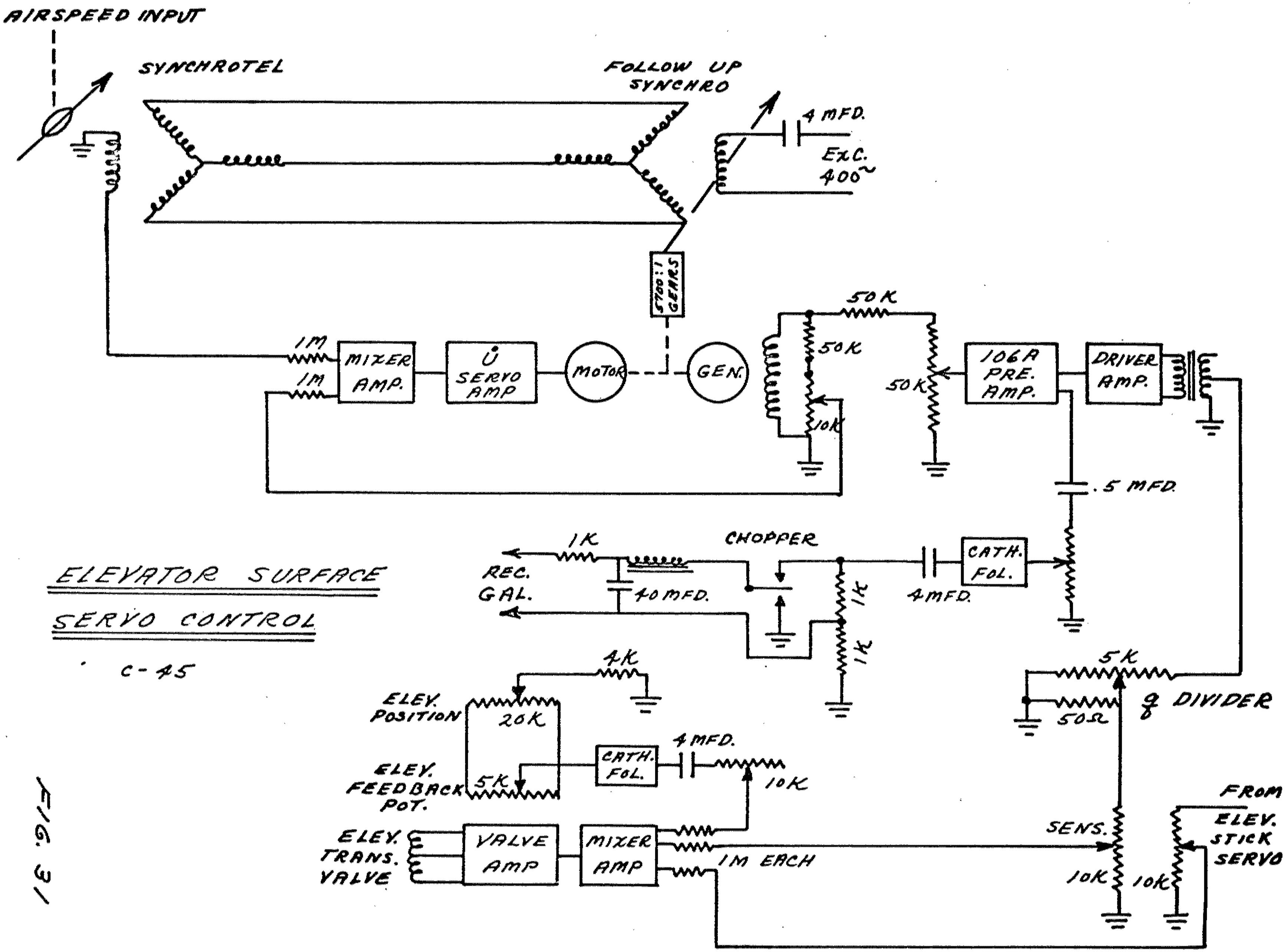


FIG. 30



ELEVATOR SURFACE

SERVO CONTROL

C-45

FIG. 31

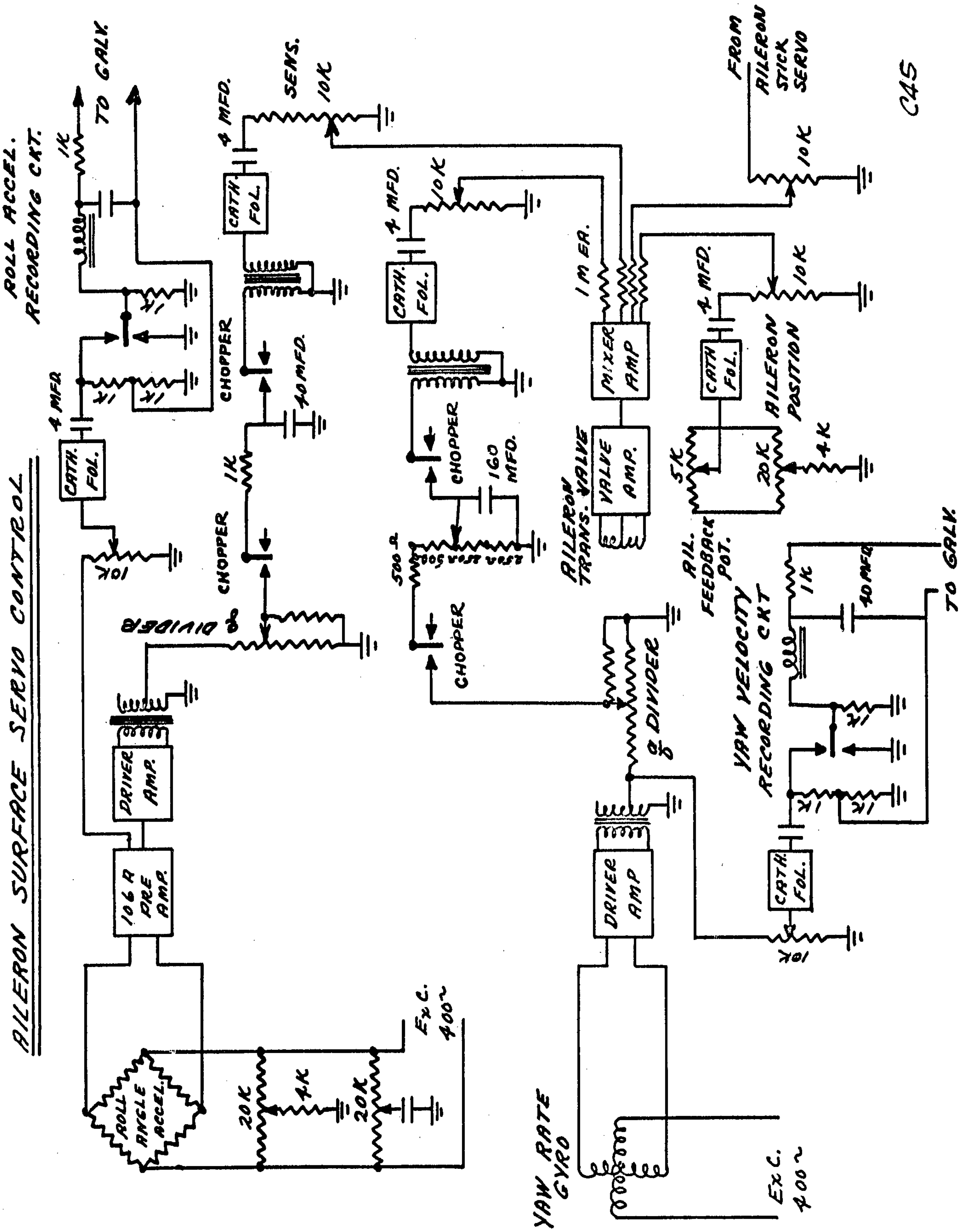
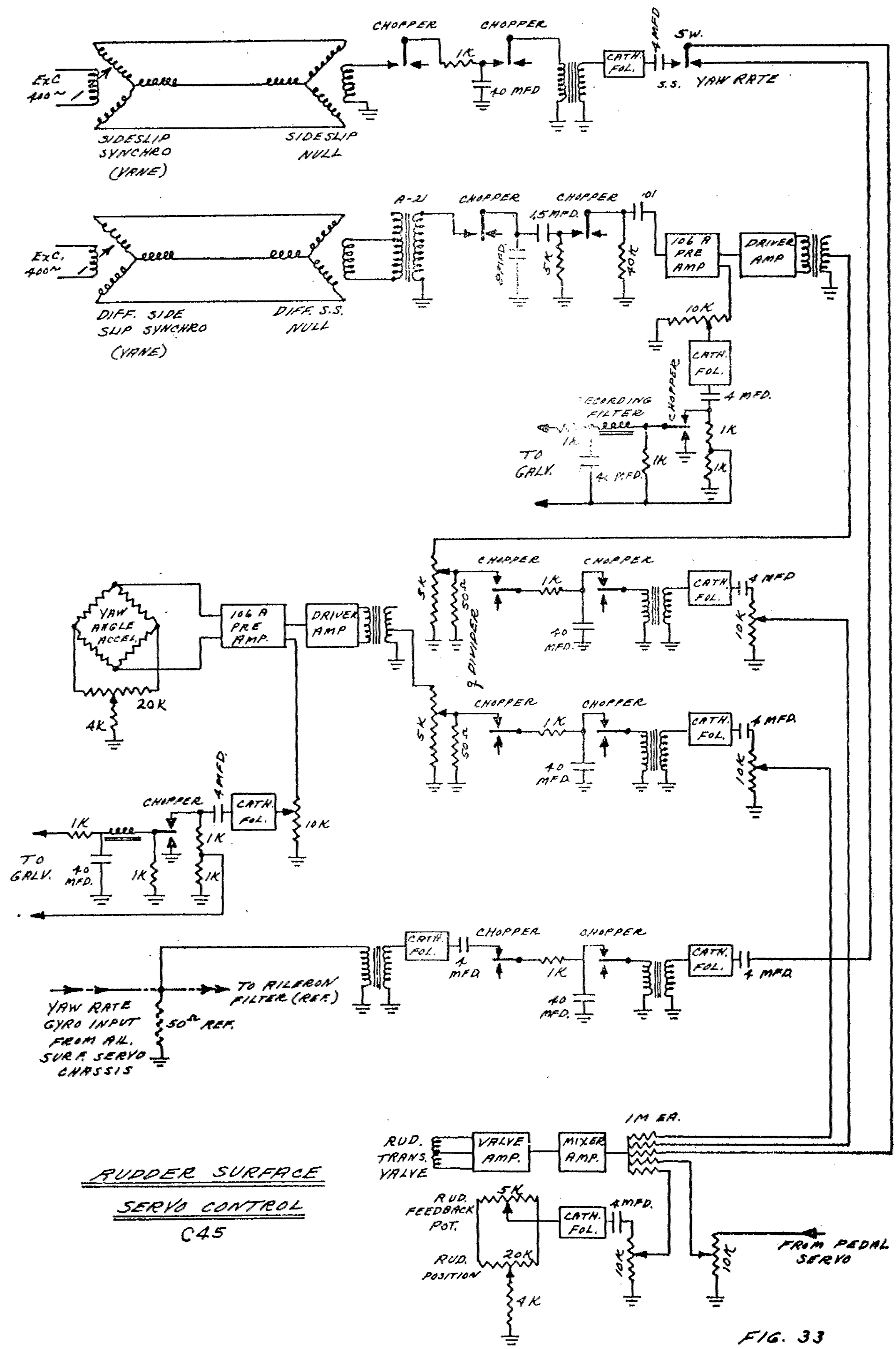

AILERON SURFACE SERVO CONTROL
ROLL ACCEL. RECORDING CKT.
TO GALV.
1K
4 MFD
CATH. FOL.
CHOPPER
10K
SENS.
40 MFD.
1/8 DIVIDER
DRIVER AMP.
106 R PRE AMP.
ROLL ANGLE ACCEL.
20K
4K
Ex C. 400~
160 MFD.
500 Ω
1 MEG.
MIXER AMP
VALVE AMP.
AILERON TRANS. VALVE
FROM AILERON STICK SERVO
AILERON POSITION
5K
AIL. FEEDBACK POT.
YAW VELOCITY RECORDING CKT
YAW RATE GYRO
C45

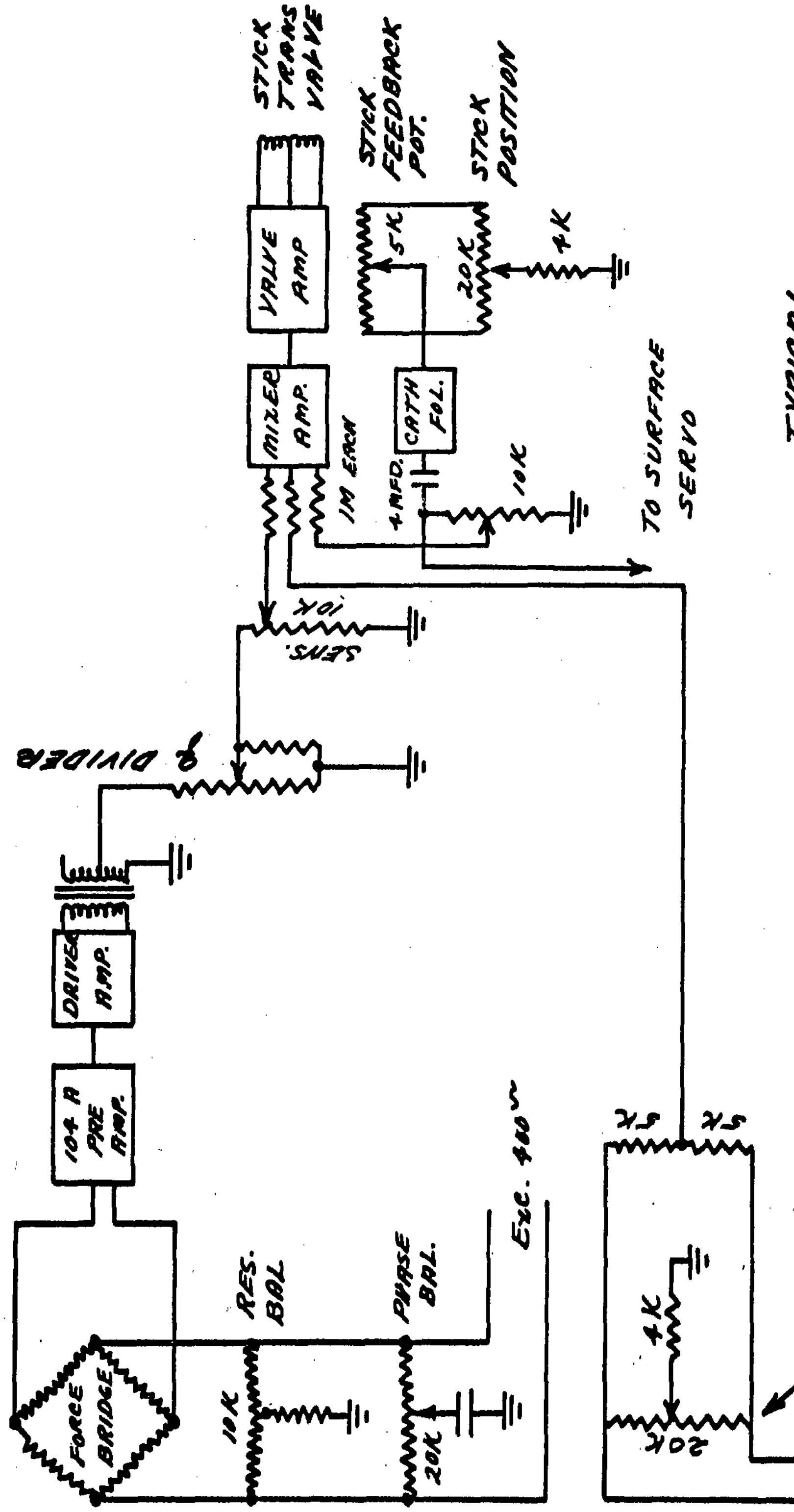
FIG. 32



RUDDER SURFACE SERVO CONTROL

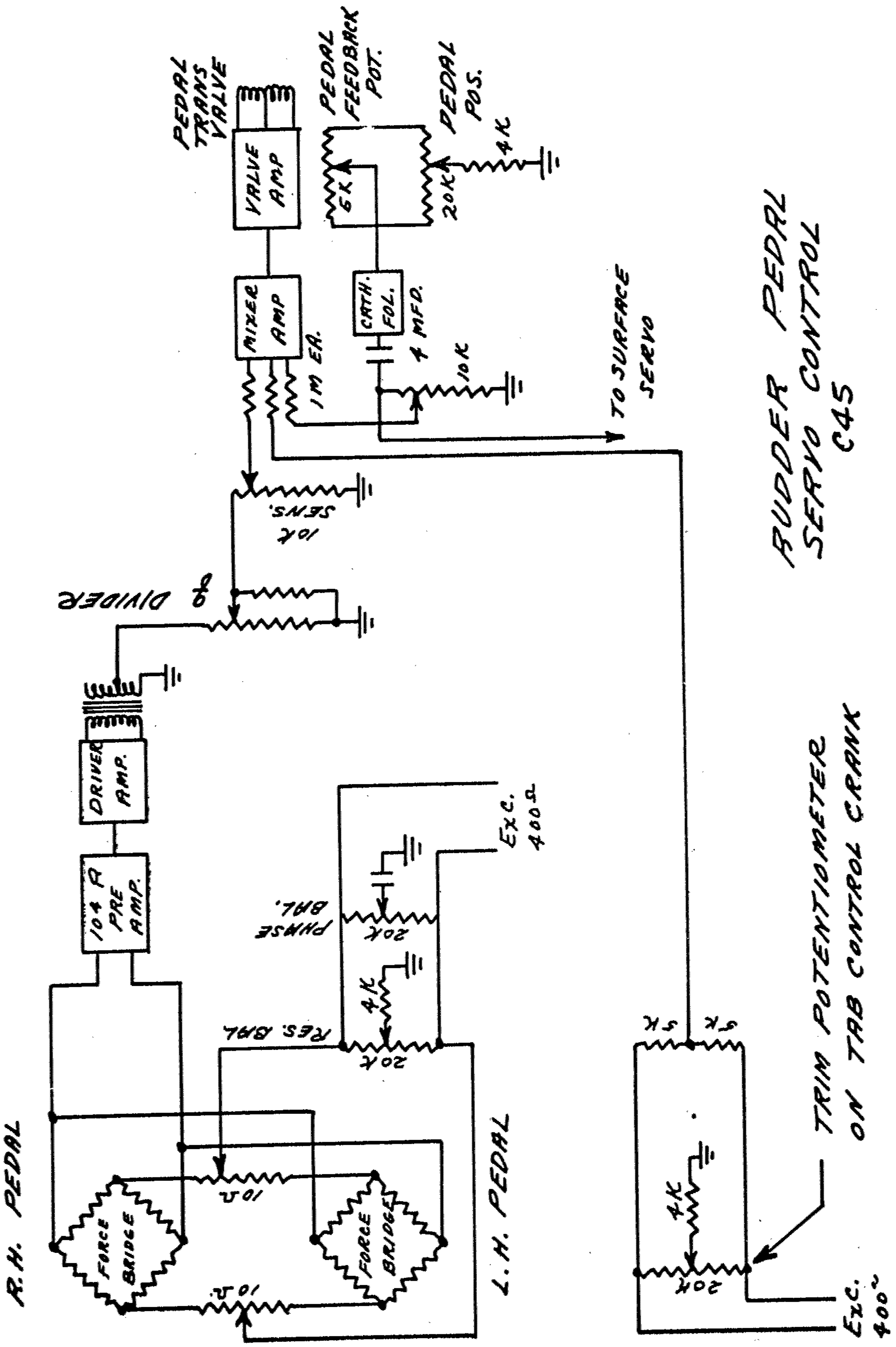
C45

FIG. 33



TYPICAL
AILERON & ELEVATOR
STICK SERVO CONTROL
CAS

FIG. 34



RUDDER PEDAL SERVO CONTROL
C45
PEDAL TRANS VALVE
VALVE AMP
PEDAL FEEDBACK POT.
6K
PEDAL POS.
4K
20K
MIXER AMP
CATH. FOL.
4 MFD.
1 M EA.
10K
TO SURFACE SERVO
10K SENS.
½ DIVIDER
DRIVER AMP.
104 A PRE AMP.
EXC. 400 Ω
PHASE BAL.
20K
4K
RES. BAL.
20K
R.H. PEDAL
L.H. PEDAL
FORCE BRIDGE
FORCE BRIDGE
10 Ω
10 Ω
5K
5K
4K
20K
TRIM POTENTIOMETER ON TAB CONTROL CRANK
EXC. 400~

FIG. 35

C-45 F

YAW RATE TO AILERON FILTER FREQUENCY RESPONSE

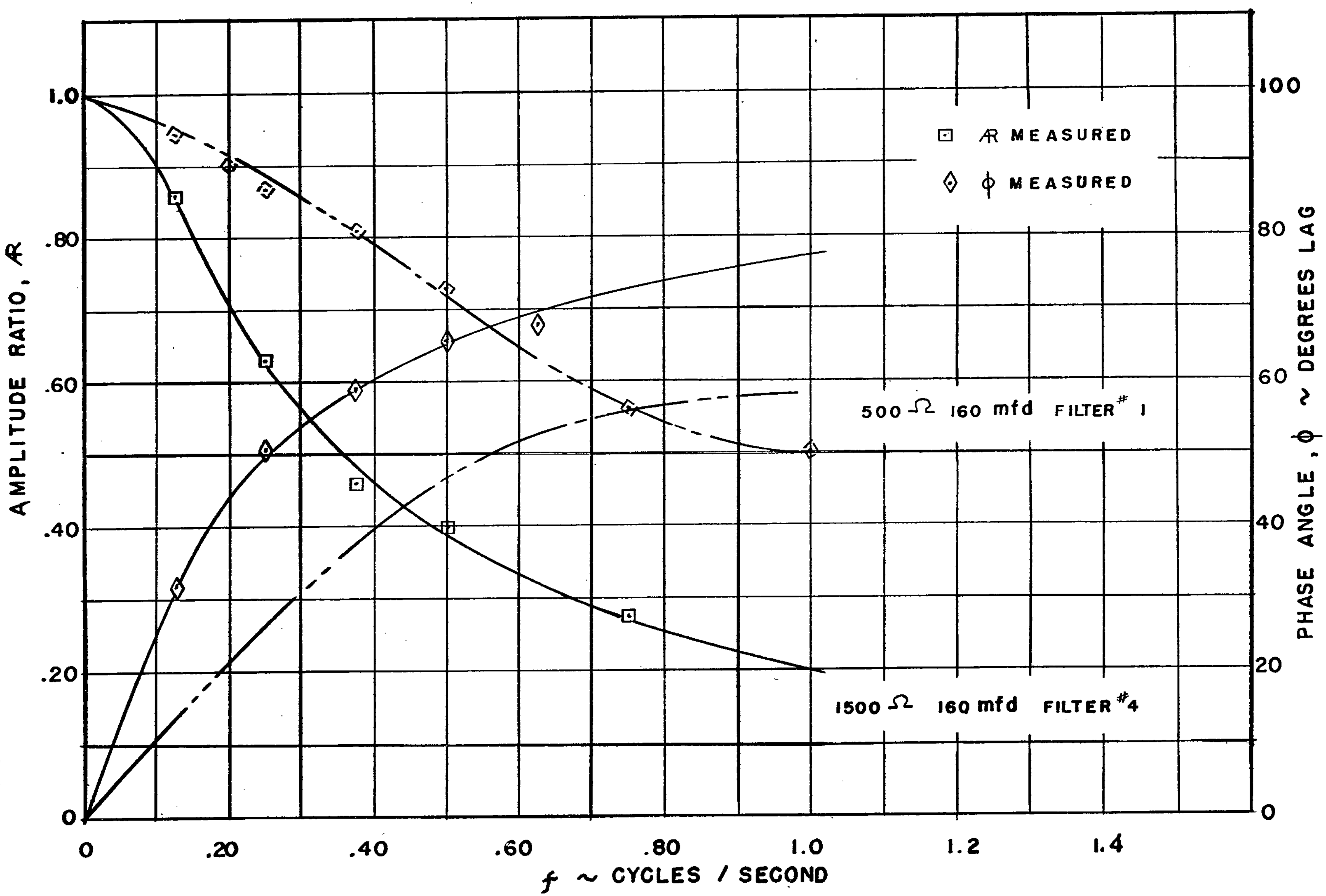


FIG. 36

CONTROL CIRCUIT FILTER FREQUENCY RESPONSE ~ C-45F

DIFFERENTIATED SIDESLIP, SIDESLIP, YAW VELOCITY & ANGULAR ACCELERATIONS

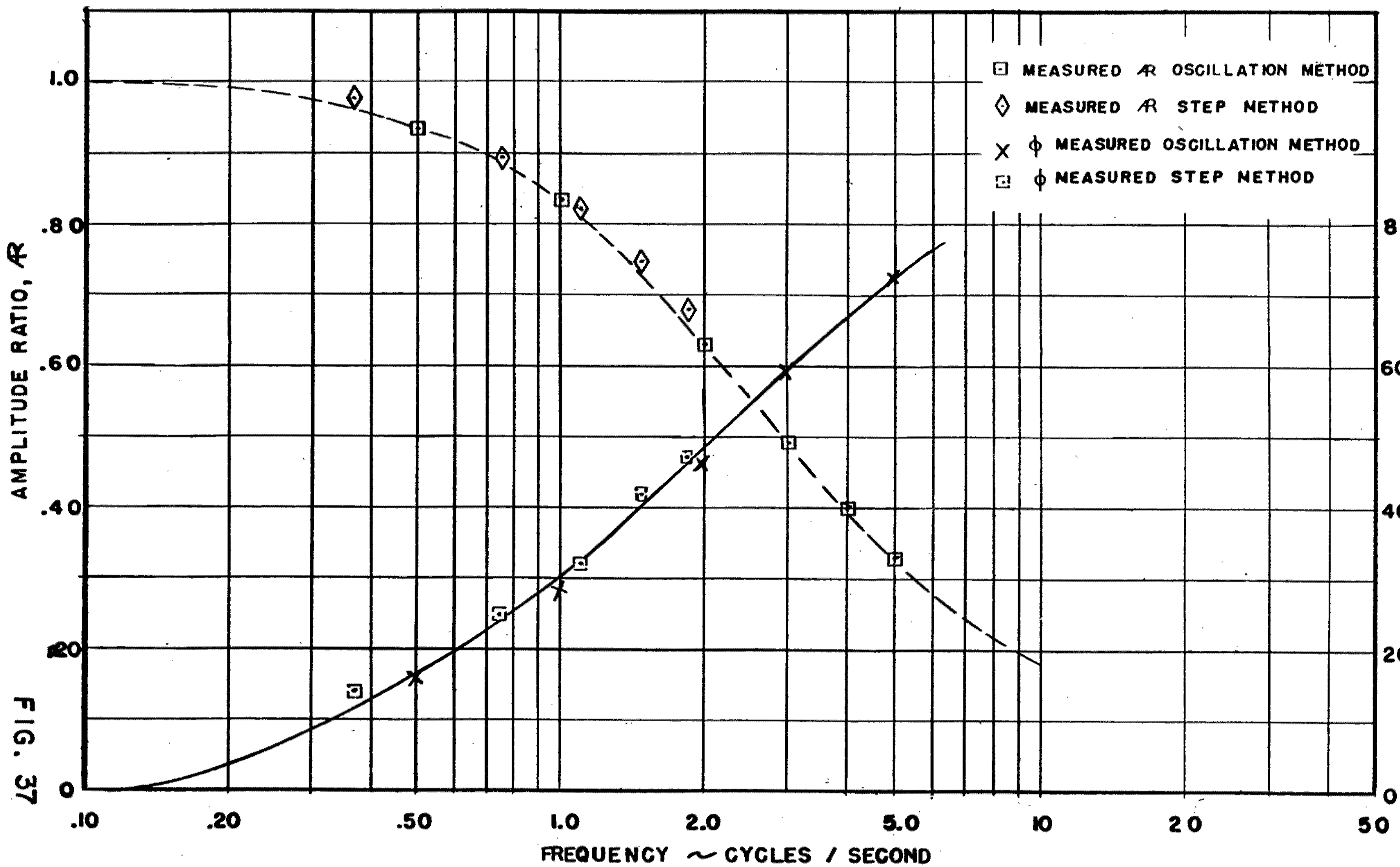


FIG. 37

C - 45F

RECORDING FILTERS FREQUENCY RESPONSES

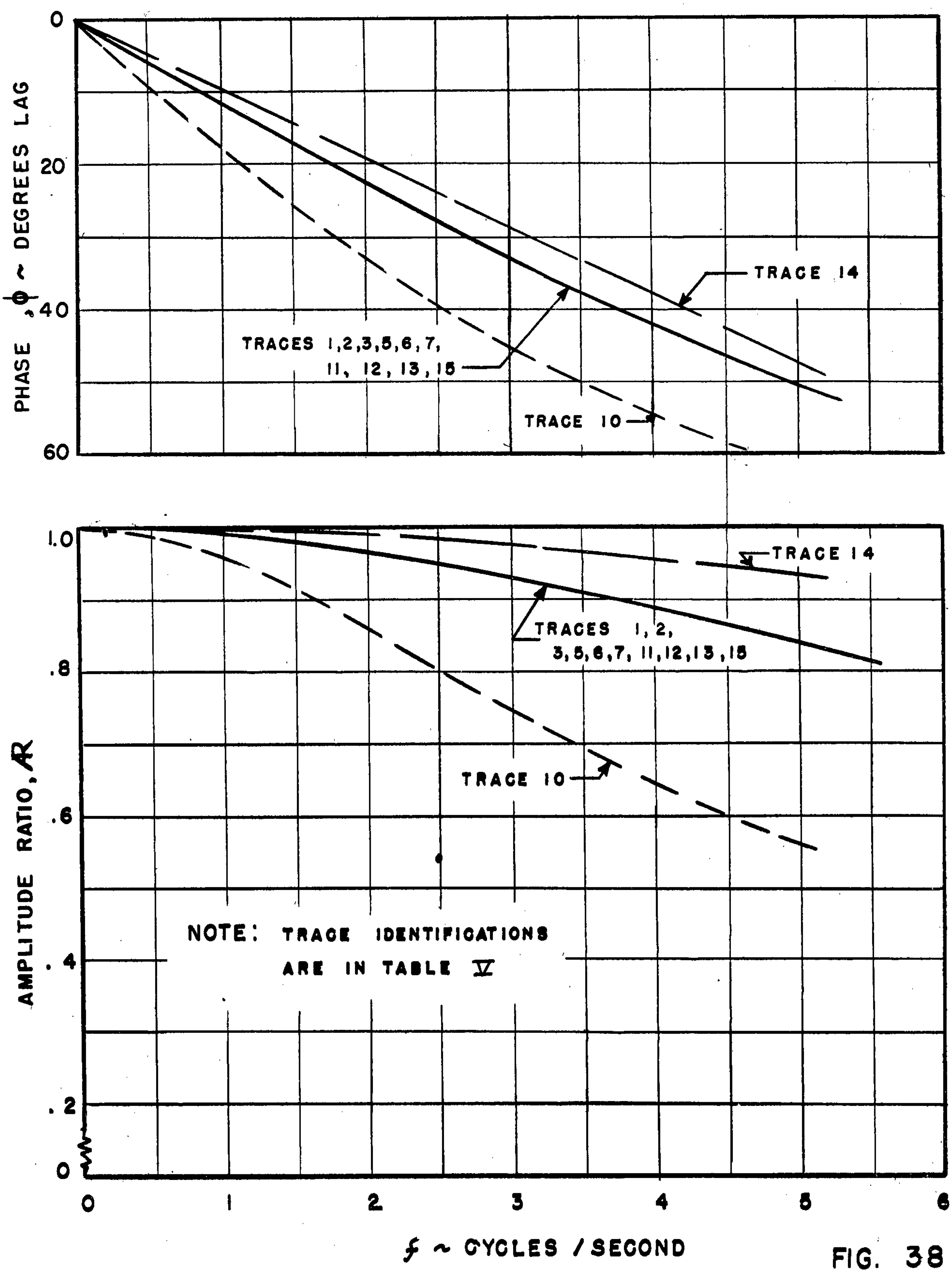


FIG. 38

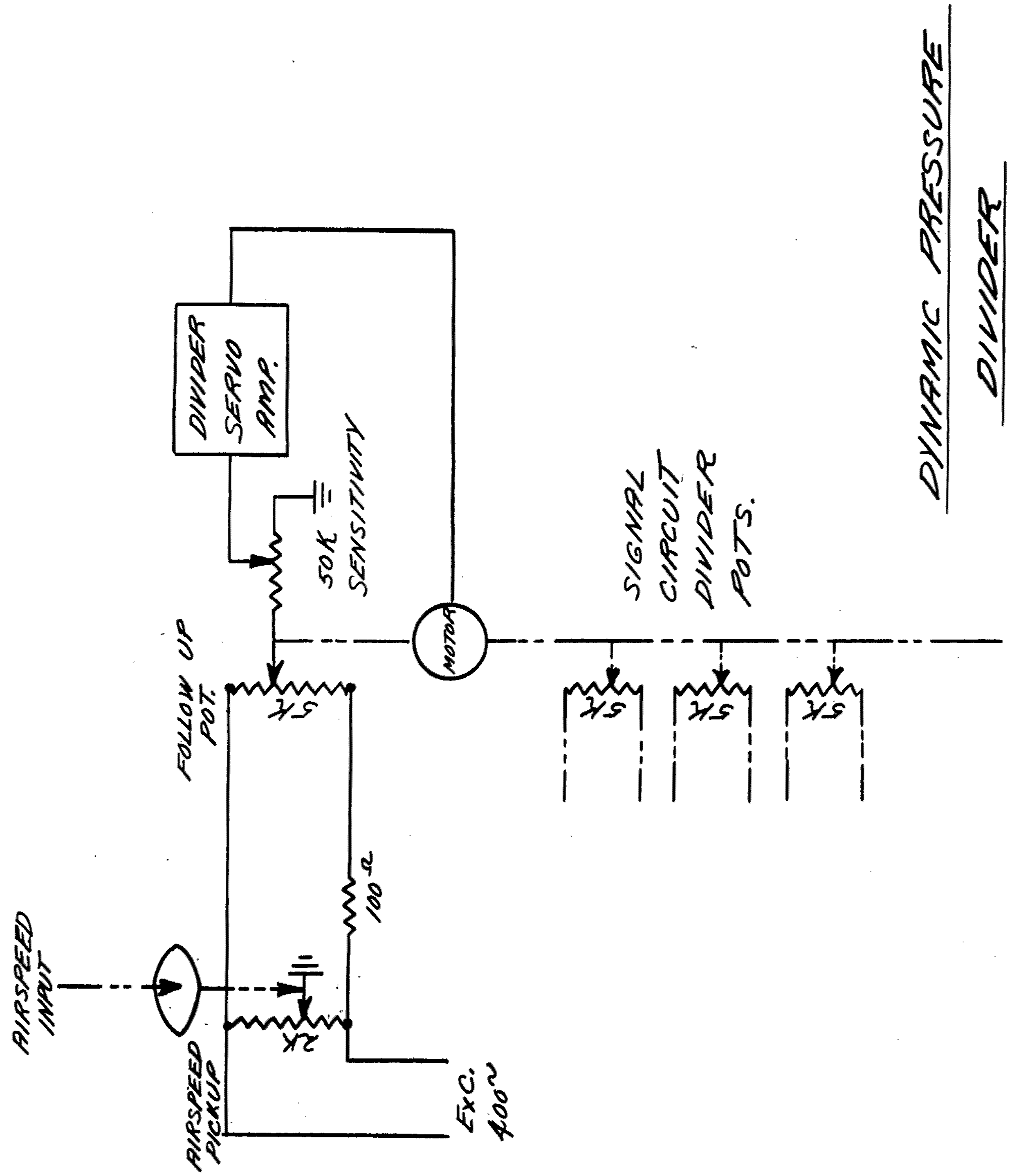

DYNAMIC PRESSURE
DIVIDER
DIVIDER SERVO AMP.
50K
SENSITIVITY
MOTOR
FOLLOW UP POT.
5K
5K
5K
5K
SIGNAL CIRCUIT DIVIDER POTS.
100 Ω
AIRSPEED INPUT
AIRSPEED PICKUP
2K
EXC. 400~

FIG. 39

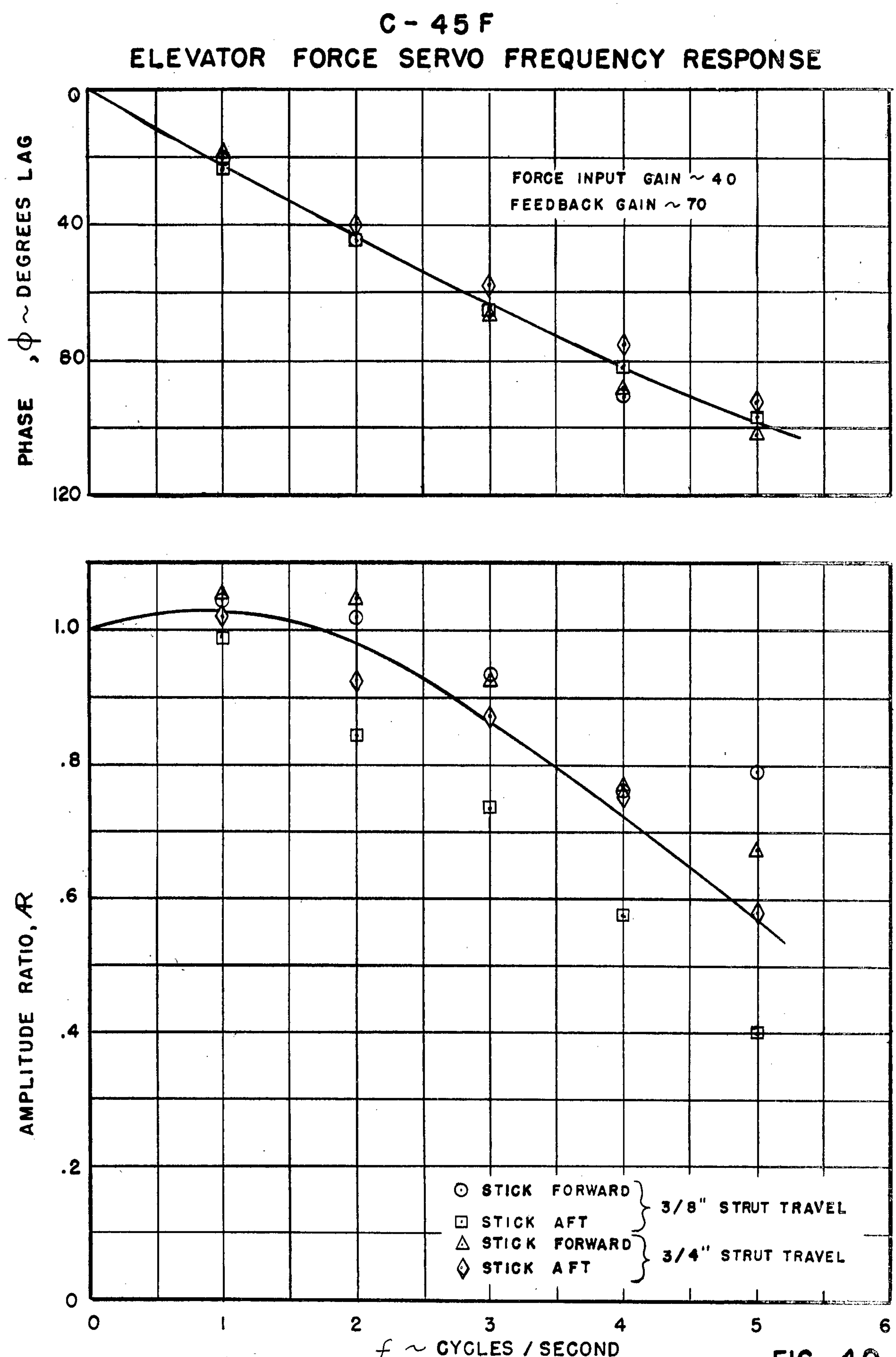

C - 45 F
ELEVATOR FORCE SERVO FREQUENCY RESPONSE
FORCE INPUT GAIN ~ 40
FEEDBACK GAIN ~ 70
PHASE , φ ~ DEGREES LAG
0
40
80
120
AMPLITUDE RATIO, AR
1.0
.8
.6
.4
.2
0
STICK FORWARD
STICK AFT
3/8" STRUT TRAVEL
STICK FORWARD
STICK AFT
3/4" STRUT TRAVEL
0
1
2
3
4
5
6
f ~ CYCLES / SECOND

FIG. 40

C- 45F

AILERON FORCE SERVO FREQUENCY RESPONSE

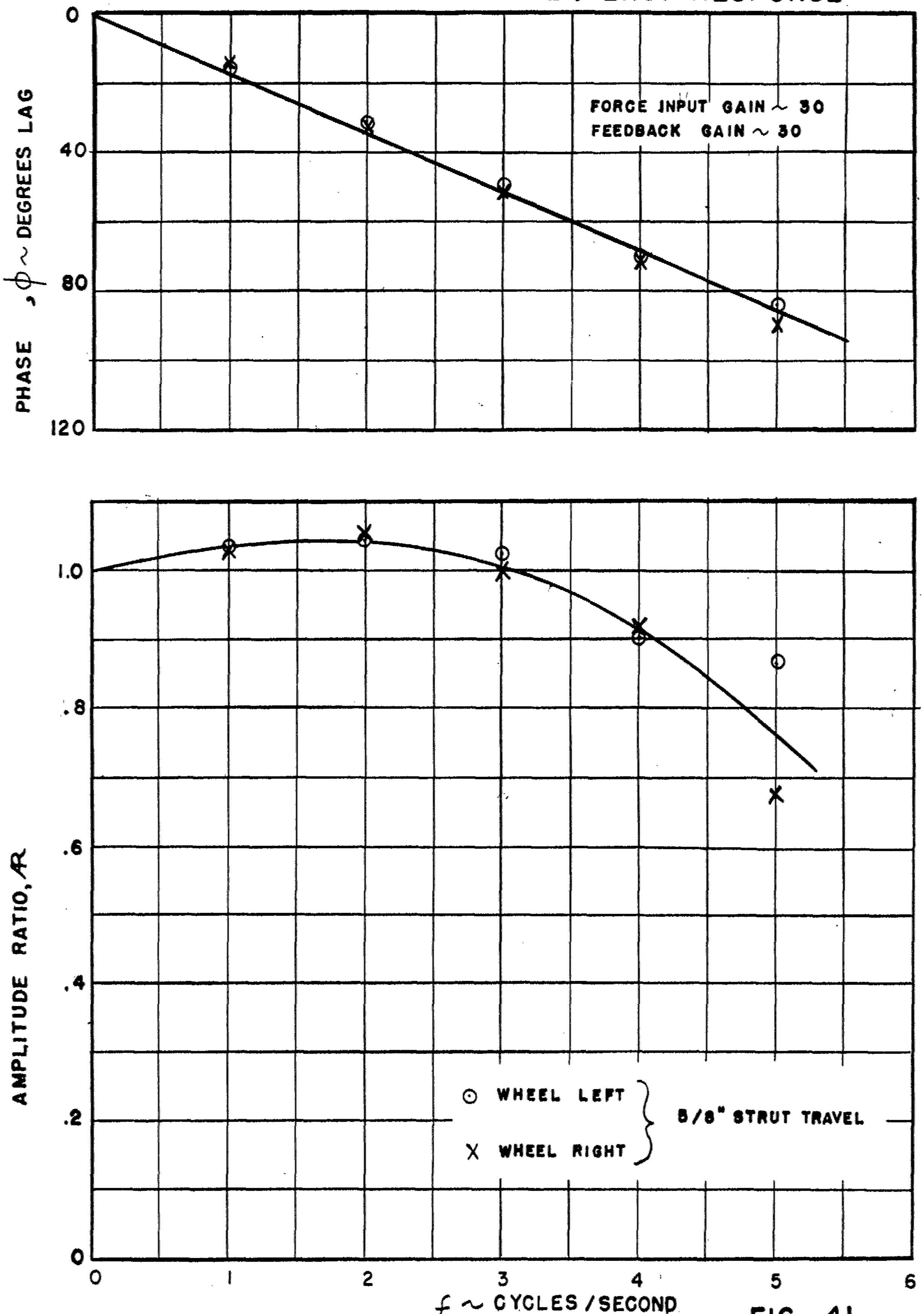


FIG. 41

C - 45 F

RUDDER FORCE SERVO FREQUENCY RESPONSE

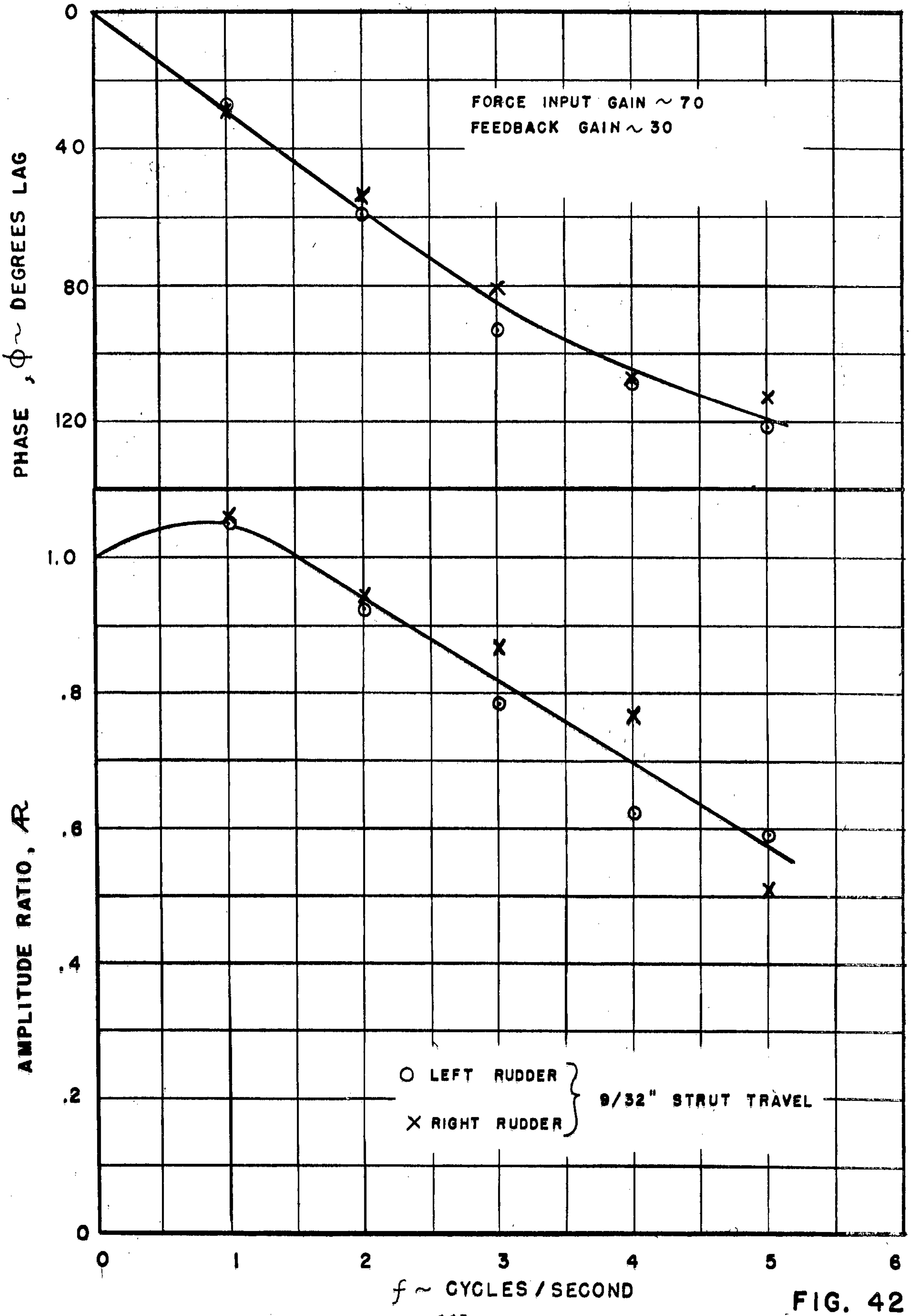


FIG. 42

C-45F
ELEVATOR SURFACE SERVO FREQUENCY RESPONSE

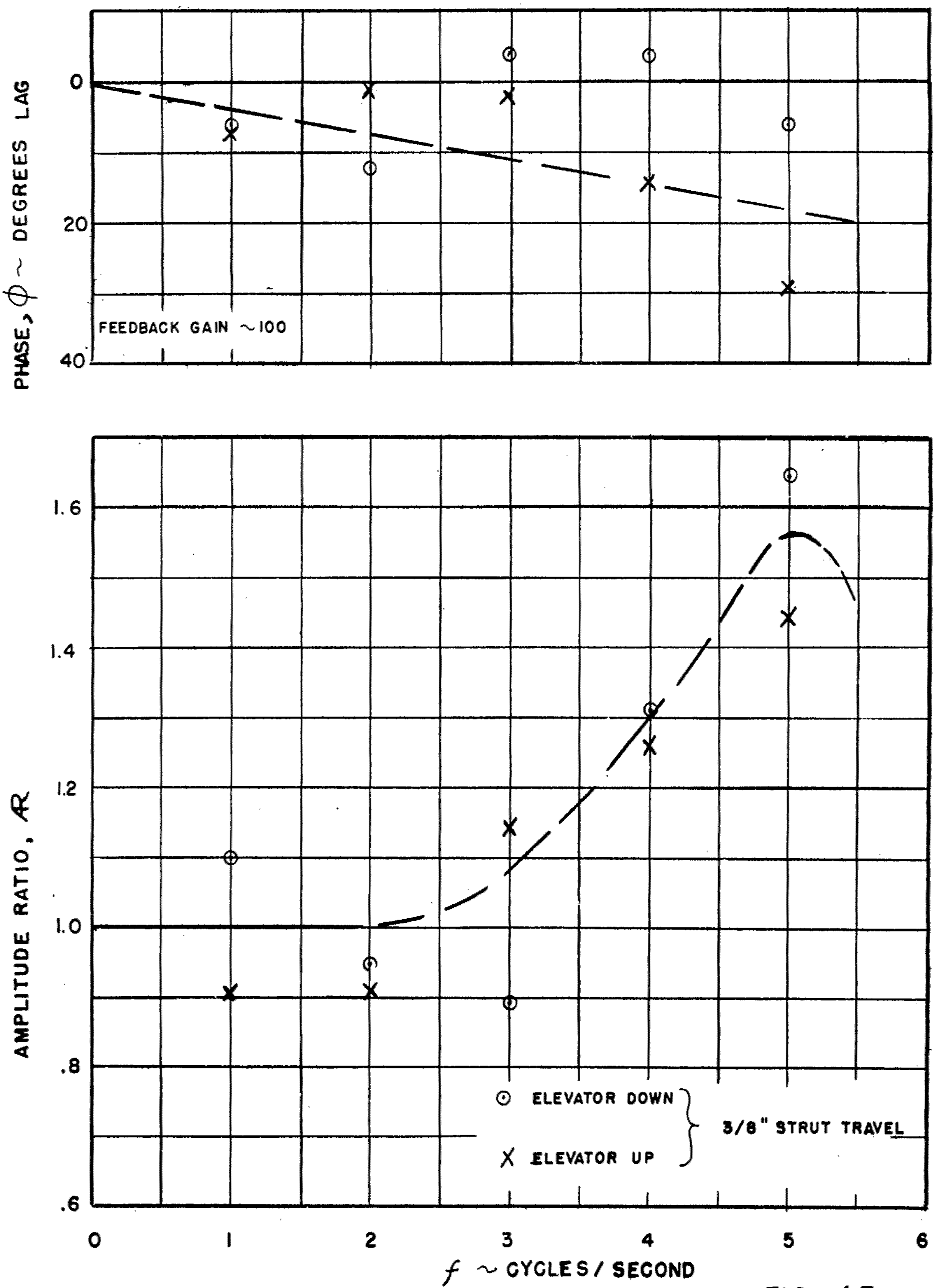


FIG. 43

C - 45F

AILERON SURFACE SERVO FREQUENCY RESPONSE

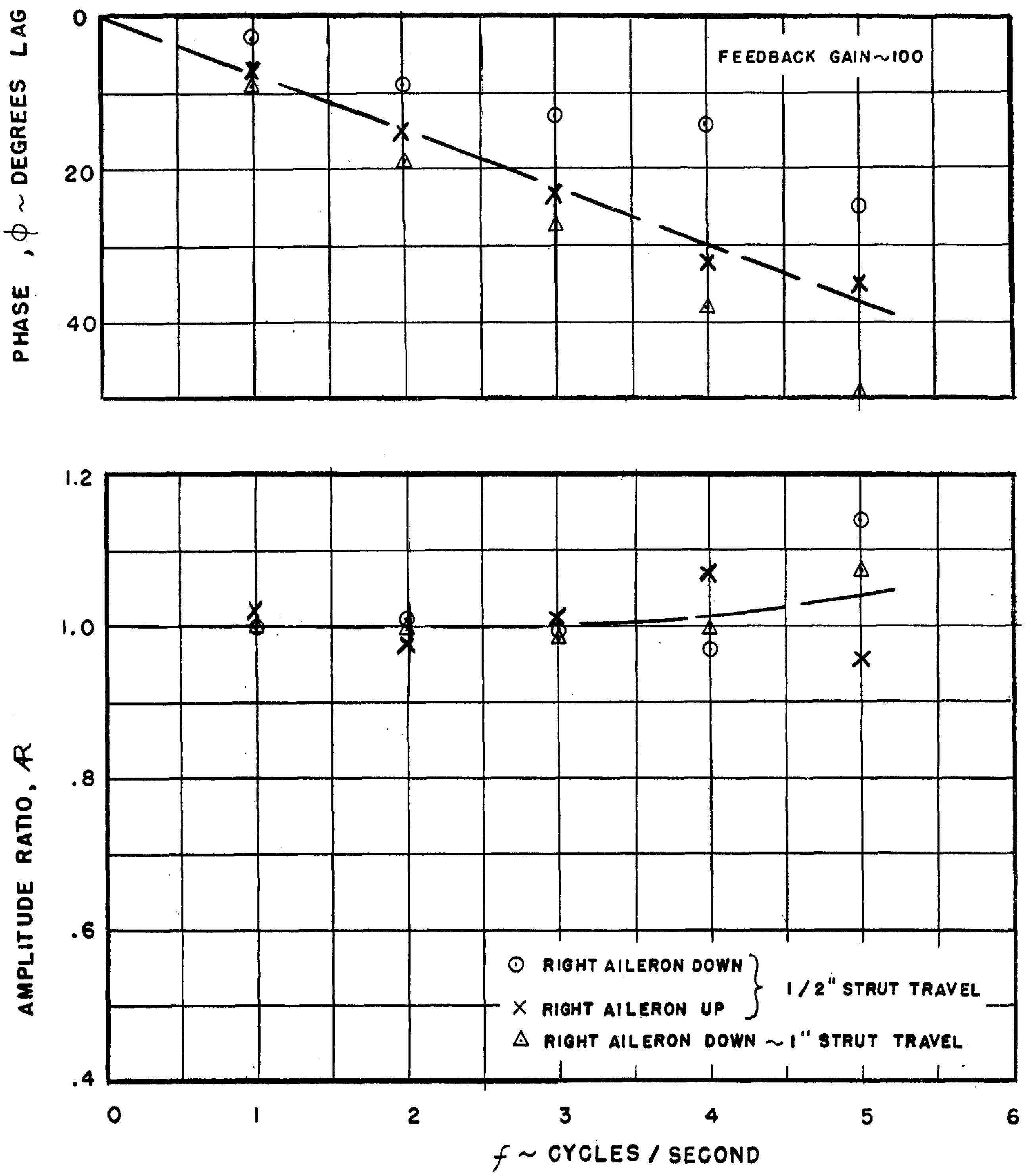


FIG. 44

C-45 F

RUDDER SURFACE SERVO FREQUENCY RESPONSE

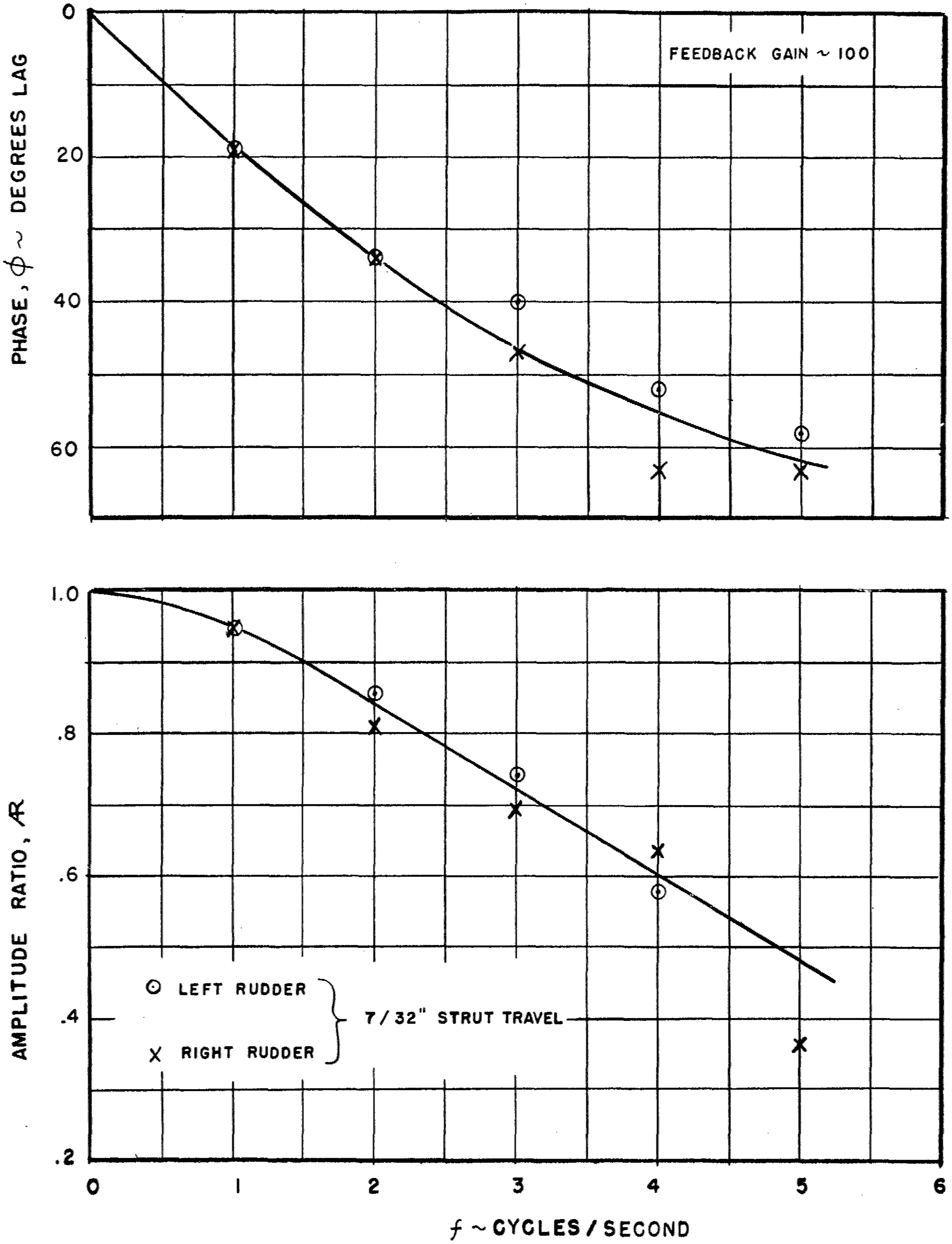


FIG. 45

C - 45 F

DYNAMIC PRESSURE DIVIDER

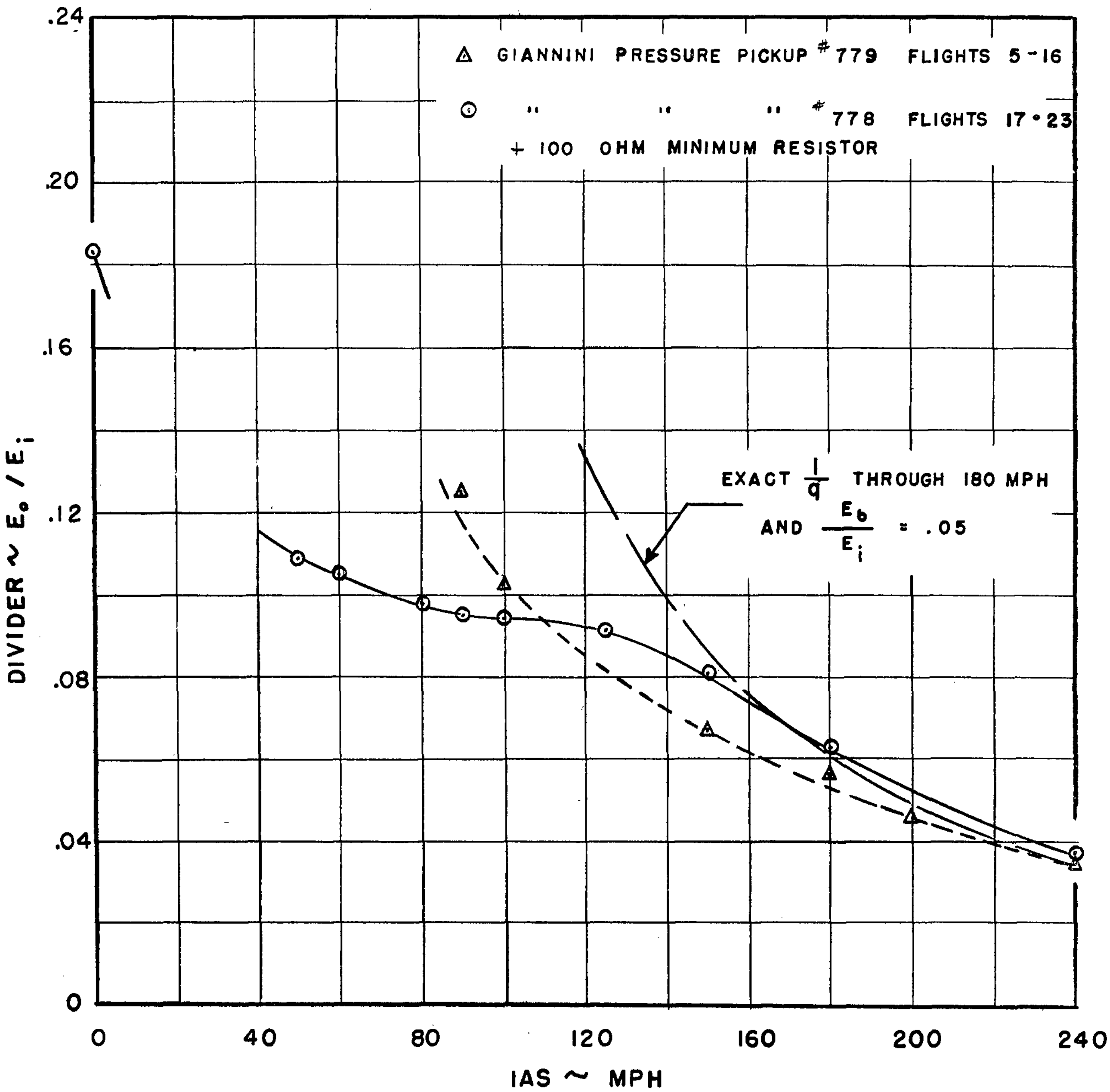


FIG. 46

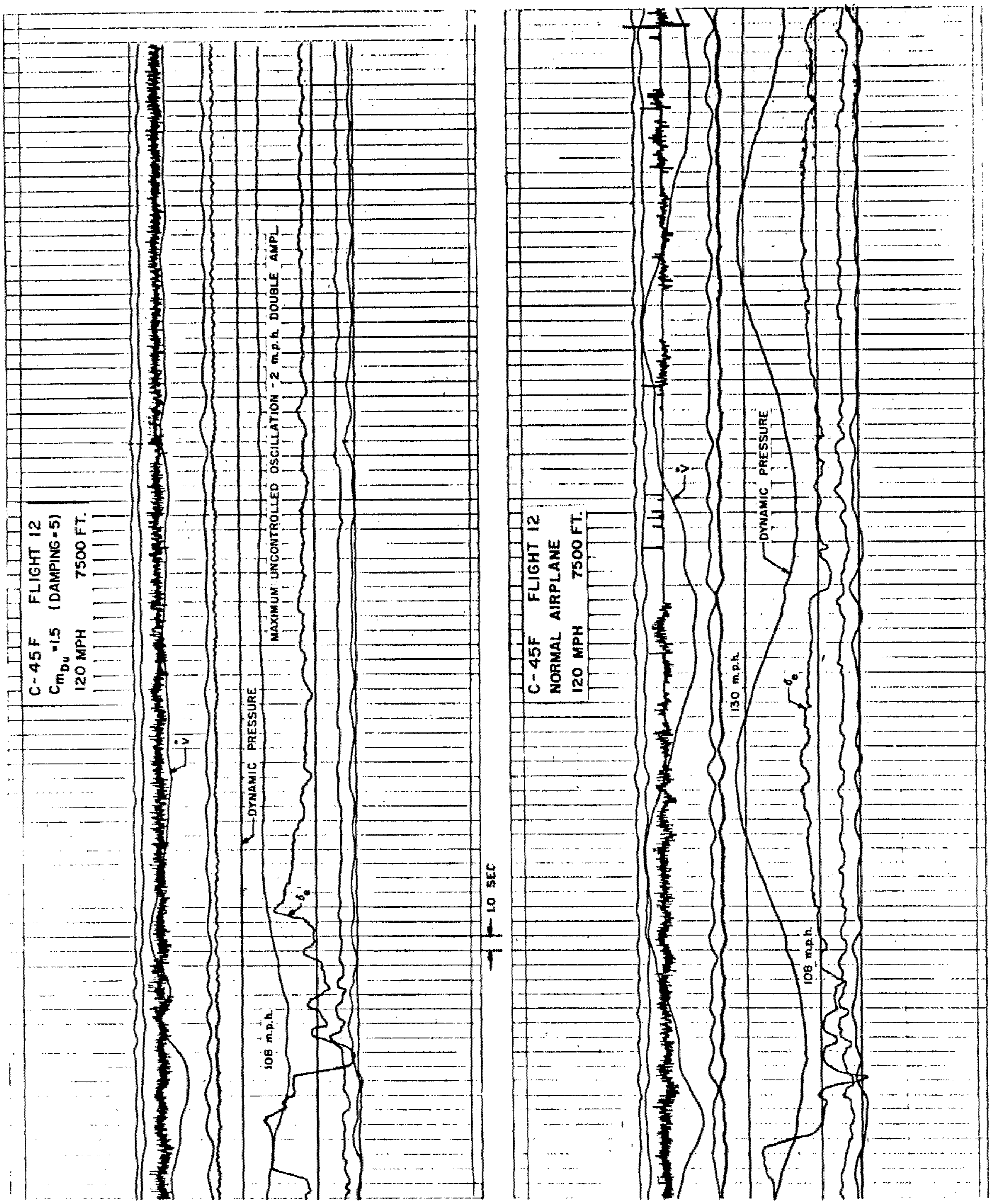

C-45F FLIGHT 12
$C_{m_{Du}}$=1.5 (DAMPING=5)
120 MPH 7500 FT.
DYNAMIC PRESSURE
MAXIMUM UNCONTROLLED OSCILLATION -2 m.p.h. DOUBLE AMPL.
108 m.p.h.
1.0 SEC
C-45F FLIGHT 12
NORMAL AIRPLANE
120 MPH 7500 FT.
DYNAMIC PRESSURE
130 m.p.h.
108 m.p.h.

Fig. 47

DUTCH ROLL RESPONCE NORMAL AIRPLANE,

$C_{n_{D_p}} = .053, \quad C_{n_r} = -.42$

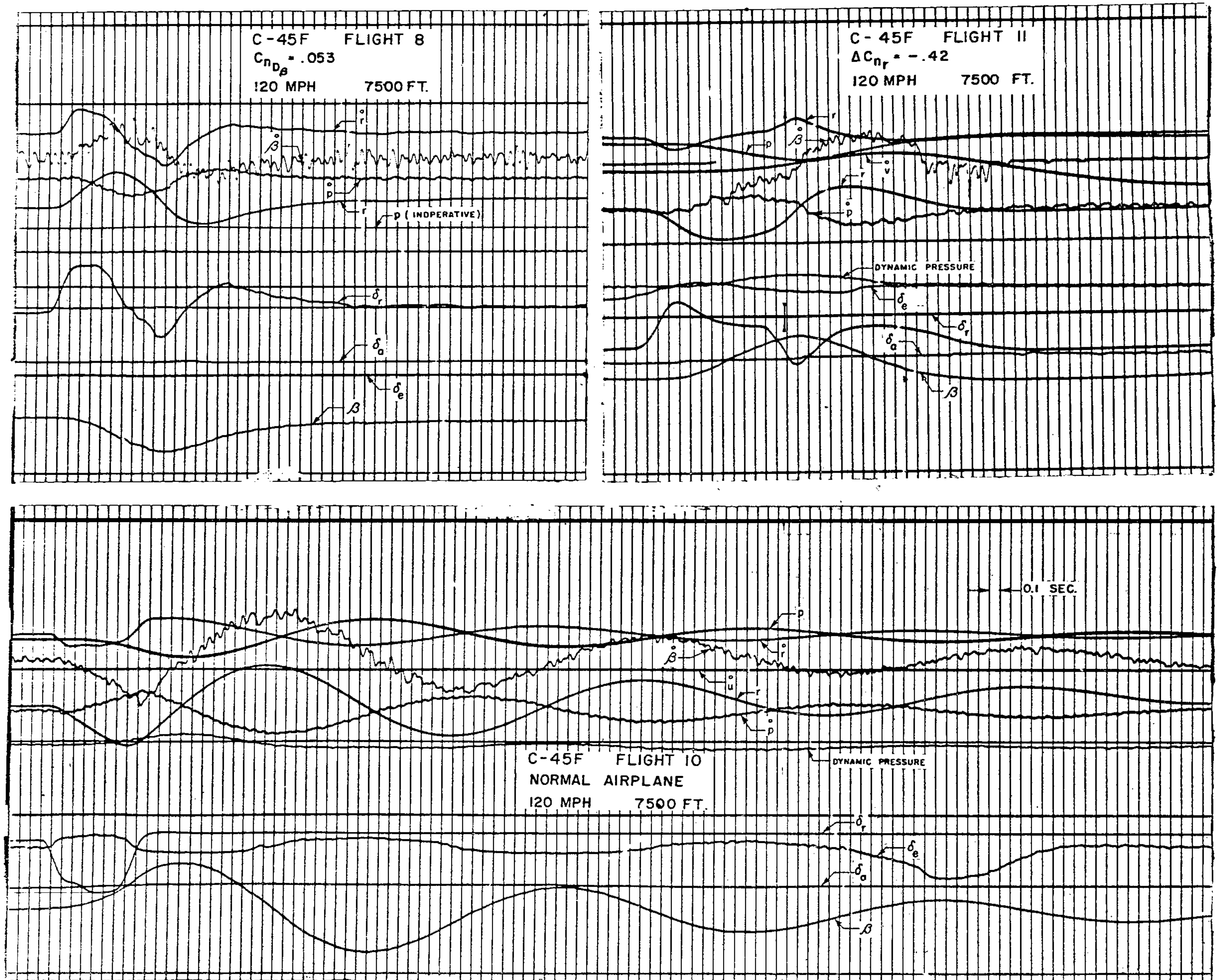


Fig. 48

C-45 DUTCH ROLL RESPONSE, $\Delta C_{\ell_r} = -30$, $\Delta C_{n_\beta} = 05$

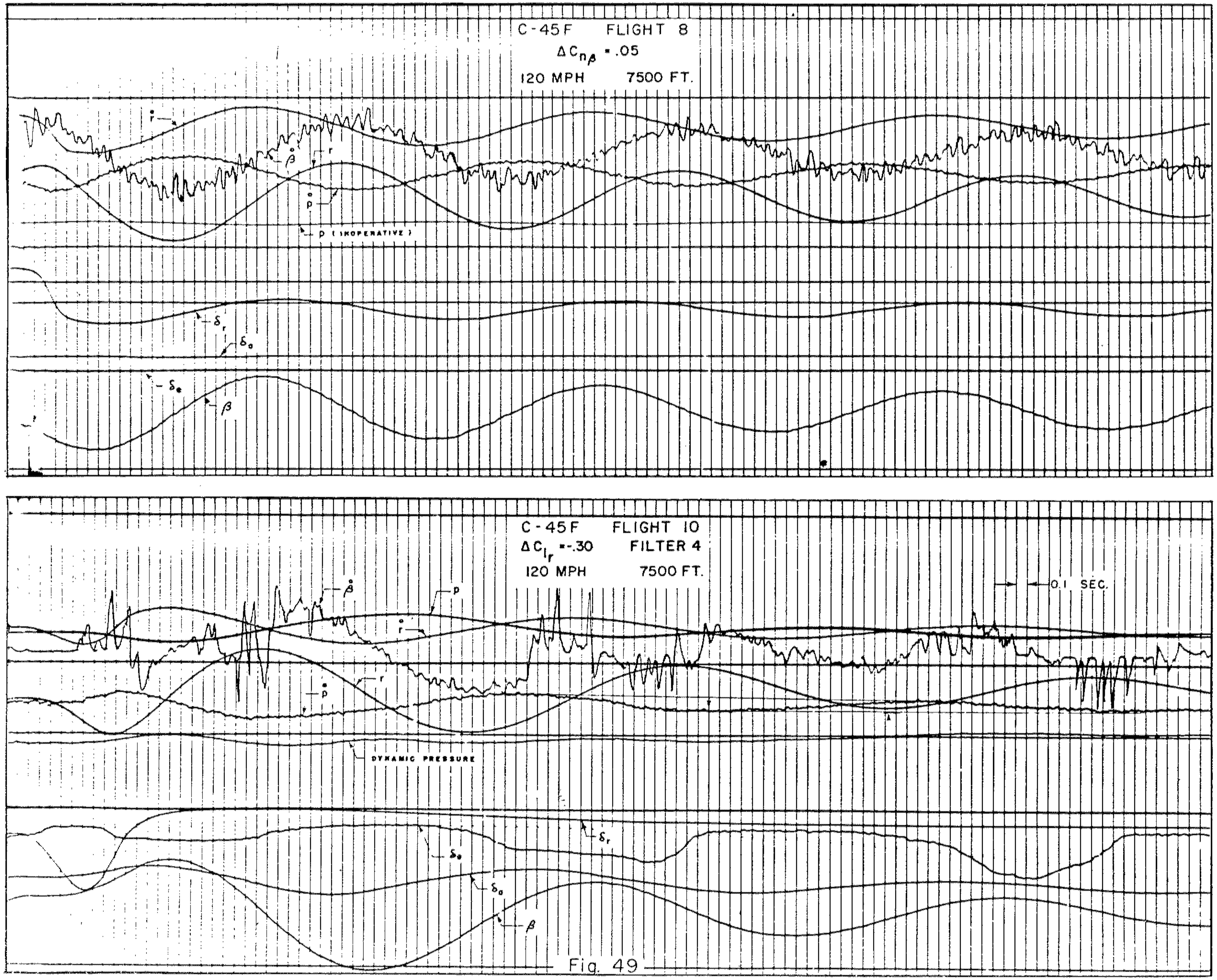


Fig. 49